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14. ABSTRACT

The space industry is always looking for new ways to improve the performance of its space systems while reducing cost and schedule. Currently, the new GPS-III satellites broadcast on multiple frequencies and host the Nuclear Detection System, however a possible alternate that uses a smaller, simpler architecture which only employs a single GPS frequency band could be used in certain scenarios. In the case that there is a gap in Earth coverage for a respective band, a single band satellite could be deployed to maintain Earth coverage and sustain constellation reliability. For a new signal that is still coming on-line, like L1C, smaller satellites could be deployed to only carry the new signal to spots where the signal is not available in the case that an urgent need develops. Conversely when a GPS satellite reaches the end of its life, it will leave a gap in coverage once it fails. Employing a small, cheap GPS satellite to fill in coverage gaps could be more cost effective and time efficient than waiting to develop, manufacture, test and launch a GPS satellite with a more traditional all-in-one architecture.

15. SUBJECT TERMS

Cubesat GPS Starlink Rapid Agile Fast E.P.I.C. Speed SMC 2.0 Global Positioning Directorate System epic speed Space and Missile Systems Center Contingency L1C L1 L2 L2C L3 L5

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ARCHITECTURE ANALYSIS FOR A RAPIDLY DEPLOYABLE GPS CONSTELLATION

by

Charles J. Salerno

A RESEARCH PROJECT

Submitted to the Faculty of the Stevens Institute of Technology in partial fulfillment of the requirements for the degree of

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1.0 Introduction

The Global Positioning System (GPS) is an integral part of sustaining the American way of life, and currently there is no back up if it goes out. If the entire constellation goes down tomorrow, then there is neither a plan nor a system available to fully restore Positioning, Navigation and Timing (PNT) information to domestic and military stakeholders of the US and its allies. Enterprises that depend on the PNT signal include commerce, aviation, power and just about every military mission. Furthermore, a surprise attack on the GPS constellation resulting loss in the PNT signal would have a crippling effect on the United States' ability to wage war, comparable its ability to wage war in the Pacific in World War II if its carrier force was present at Pearl Harbor in 1941. In the event there is a severe PNT outage, it is not okay for military space leaders to ask: "what do we do now?".

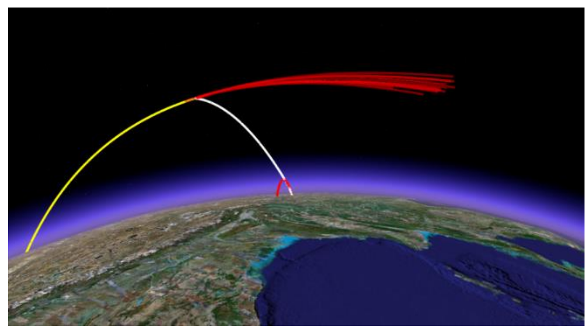


Fig 1. Depiction of the 2007 Chinese ASAT Test (2)

This nightmare scenario is not just rooted in tinfoil-hat paranoia. According to the *Final Report on Organizational and Management Structure for the National Security Space Components of the Department of Defense*: "Some new Russian and Chinese [Anti-Satellite] ASAT weapons, including destructive systems (see Fig 1), will probably complete development in the next several years … Both countries are advancing directed energy weapons technologies for the purpose of fielding ASAT systems ... [and] continue to conduct sophisticated on-orbit satellite activities such as rendezvous and proximity operations, which are likely intended to test dual-use technologies with inherent counterspace functionality." The report then identifies "Alternate positioning, navigation, and timing for a GPS-denied environment" as a focus of capability development in the Department of Defense Space Vision. Loss of the PNT signal is a clear concern for the United States (1).

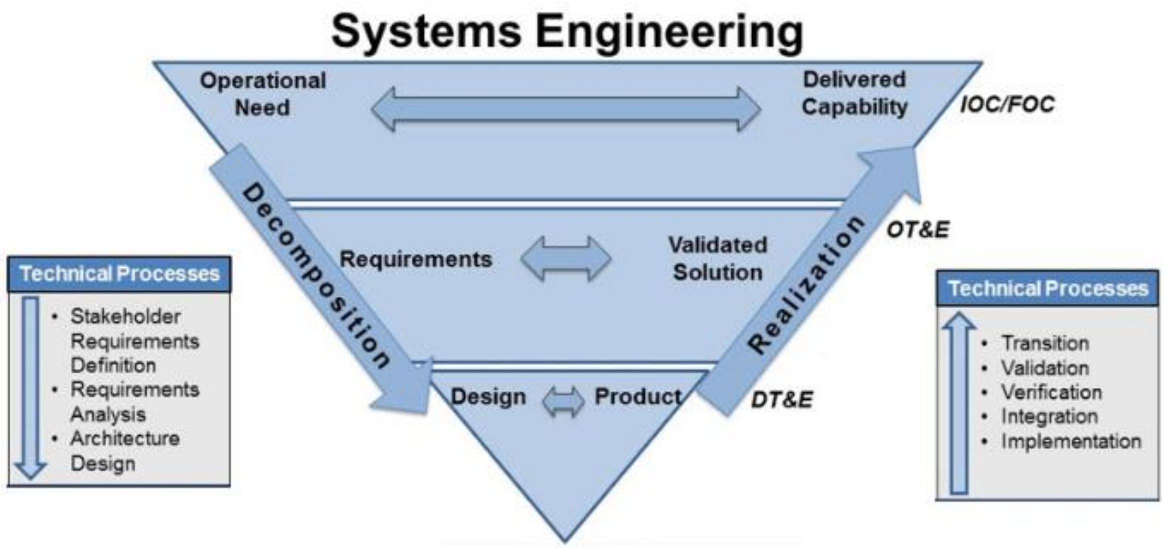
The problem addressed here is very much a second Space Race of Acquisition. Russia and China are developing ASAT capabilities, and if they fully develop these capabilities before the US develops a capability to safe its PNT system, then the ability for the US to wage war is at risk. However, the very real programmatic constraints of cost, schedule and performance must be considered when addressing the acquisition of a system of the proposed scale. Constructing an entire GPS constellation in the same way as the current one would take a very large amount of time and money. Not only is the investment in acquiring a completely new GPS constellation astronomical, only so many space vehicles (SVs) can be launched at one time. As a result, alternate ways of providing PNT must be seriously considered. If the current GPS constellation gets quickly taken out in a decisive attack, then a need for a replacement is immediate. The solution system will not be useful to anyone if it is needed while it is slowly deployed, halfway through development--or worse, nonexistent … therefore the solution must have the ability to be rapidly acquired and as well as rapidly deployed, if, heaven forbid, it is ever needed.

Small satellites, such as the ALTAIR in Figure 2, are carving out their niche in the space industry and can applied to the problem of acquiring alternate sources of PNT. Technology such as more efficient solar cells, batteries, propulsion systems and smaller electronics are allowing economical alternatives to traditional larger space vehicles. As satellites get smaller and contain less components, the production time will decrease since every component in the space industry is prone to hyper precise inspections and qualifications. Additionally, small satellites give the opportunity to use ride sharing on CubeSat deployers on commercial launches and the opportunity to be used on smaller launch vehicles (LV) to cut down on costs and deployment time. Since LV selection is a major driver in cost and schedule to putting any satellite on orbit, having systems that can utilize alternatives is huge in the space industry. The goal of this paper is to use the systems engineering process to propose an architecture for what a possible solution will look like: a rapidly acquirable and deployable small satellite constellation capable of filling in PNT coverage gaps using current or “very near horizon” technologies.

Fig 2. ALTAIR Pathfinder deploys from NanoRacks CubeSat Deployer (3)

2.0 Systems Engineering Methodology

Figure 3. The DAU Systems Engineering Process (4)



This systems engineering study will first begin with the classic "V", depicted in Figure 3 taken from the Defense Acquisition Guidebook by Defense Acquisition University (4). Systems engineering is used by the Department of Defense to identify an operational capability gap and deliver a system that fulfills that gap. The process starts by identifying a capability gap, which is a need that when fulfilled will add value to the warfighter and key stakeholders. For example, a hypothetical capability for the Air Force would be attack ground targets in faraway access aerial denied areas which could then lead to the development of a new long range stealth bomber. A new long range stealth bomber would not be developed for the sake of developing a new long range stealth bomber. Once an operational need is identified, requirements for what the system must do have to be written (see Table 5). After the requirements are finished, the architecture for what the system to fulfil the requirements would look like can be developed. Architecture precedes detailed design since it defines what functions individual components must fulfil, a critical step for delivering the right product. Identifying an operational need, writing requirements and creating an architecture is known by DAU as "Decomposition" and will be tailored to this research (4).

For this research, first the operational deficiency or "need of the customer" is identified. Then, active and passive stakeholders that are affected by the need/opportunity are identified along with their expectations. Active stakeholders are those that will directly interact with the system and passive stakeholders influence the system while not directly interfacing with it. An example of a passive stakeholder would be a regulatory agency like the Federal Communications Commission. Then, the system's context is determined. A context diagram and use case diagram can be used to show how the system interacts with other systems and stakeholders. During this phase, different concepts are selected and compared, however the concept for this research was pre-selected as a small satellite operating in Low Earth Orbit (LEO) to be an exercise in space systems engineering much like the Firesat example in Applied Space Systems Engineering. In a purist systems engineering process, different concepts for PNT signal distribution such as land based stations or high altitude balloons would also be considered.

After the context in which the system will be used is defined, a Concept of Operations (CONOPS) and requirements will be written to define the operation of the system. Included in the CONOPS will be a mission timeline, a use case diagram and an OV-1. System requirements will be written by being broken up into functional and non-functional requirements derived from the CONOPS and stakeholder expectations. According to Applied Space Systems Engineering (16): functional requirements typically map to stakeholder expectations of functions the system must perform while non-functional requirements map to characteristics of the system such as performance, availability, cost, mass etc…

Finally, after doing the systems engineering prework, technical analysis on the proposed space system's physical architecture will be presented. First, an analysis of the required orbit will be conducted to determine where the proposed space system will fly to in order to drive component sizing. The orbital analysis will be supported graphically by Satellite Constellation Visualization (SaVi), a free Linux based program developed by Lloyd Wood (32). After an orbital profile is determined, analysis can be conducted to determine what the subsystems of the space segment will look like. The research will skip functional architecture mapping and go straight to analysis of the physical space segment elements since author engineering judgement can be used to determine the high level physical architecture from the existing GPS system. The analysis will be supported heavily by the Space Mission Analysis and Design (SMAD) Spreadsheet v6.1 (17) as well as NASA's 2018 report on the State of the Art of Small Spacecraft Technology (20). Once the top level space segment architecture has been determined, a cost estimate derived using the SMAD spreadsheet comparing the proposed system with the current GPS III satellites will be presented.

3.0 Stakeholder Analysis

The first step in the research was to determine who would be affected by loss of the GPS constellation. "Everyone who uses a smartphone" would be a true statement, however a deeper understanding of the current GPS system is required since the contingency small satellite constellation would ideally be "plug and play" with existing infrastructure. Therefore, knowledge of who controls the existing GPS constellation and which end users require GPS to maintain safety of life is critical. Table 1 is an attempt to categorize active and passive stakeholders across the GPS enterprise from management, operations, interfaces and end users. Figure 4 shows an illustration of the ground operator to GPS system interfaces which is documented in Table 1. In Table 1, the name of the stakeholder, their relation to the GPS system (active vs passive), a top level description of the who the stakeholder is and what they expect from the GPS system is tabulated. Stakeholder expectations where derived through research of literature, websites, official reports and the author's engineering judgement.

Figure 4. GPS Ground Control Segment (6)

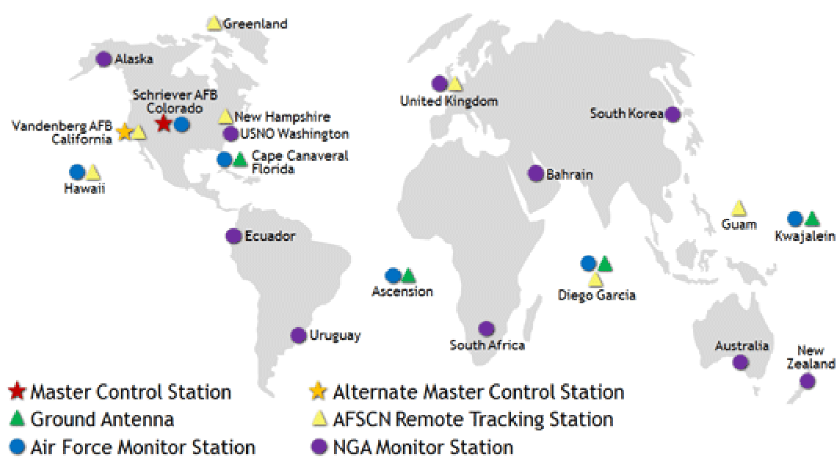


Table 1. GPS Stakeholders

Stakeholder	Type	Description	Expectation
USAF Global Positioning Directorate	Passive, Sponsor	Provide life cycle program management for GPS systems. Located at the Space and Missile Center, Lose Angeles AFB, CA. Directorate will align with Center objectives. Center policy will align with national political objectives. Figure 6 shows a picture of LA AFB.	System must meet technical requirements while minimizing cost and schedule. SMC 2.0 restructures way Air Force does space acquisitions in order to foster innovation and move faster than our adversaries. Lt Gen Thompson, SMC/CC stated: "We need to be able to fight and defend ourselves in space, and we do that by deploying an agile, resilient and secure C4 [command, control, communications and computers] space architecture." (5) Meaning: Wants a robust space presence that can respond rapidly to threats. Align with POTUS directives for space resilience and vision for Space Force One of the pillars for US space leadership listed in the President's National Security Strategy and the National Strategy for Space is: "Transform to more resilient space architectures" (1) Meaning: Does not want to lose US space superiority Space Force Report identifies " Establish a Space Development Agency, a joint organization charged with rapidly developing and fielding next-generation capabilities," as an immediate step for the DoD and specifically calls for "Alternate positioning, navigation, and timing (PNT) for a GPS-denied environment," as a department developmental capability focus (1) Meaning: We need alternatives to our current GPS constellation, and we need them quickly
2nd Space Operation Squadron (2SOP)	Active	Daily upload of nav signal, monitoring, diagnostic, reconfiguration, station keeping.	Voice of customer: "We always check the health of the satellite to make sure everything is functioning correctly that way we can give the

		Master control station located at Schriever AFB, CA and the alternate control station located at Vandenberg AFB, CA	precise navigation signal" – A1C Jareo Brumsfield (6) Meaning: 2SOP must be able to interface with the satellites from their control stations to send and receive Space Vehicle (SV) data with little delay or down time
1st Space Operation Squadron (1SOP)	Active	Prelaunch, launch, orbit insertion, anomaly resolution and monitoring. Schriever AFB	Same as 2SOP. Must have seamless transfer of control authority due to use of different ground system (6)
National Geospatial Intelligence Agency	Active	Provide 10 monitoring stations under Legacy Improvement Initiative (L-AII)	SV signals must interface with NGA monitor stations (6)
Air Force Satellite Control Network	Active	Provides 7 ground antenna to upload commands, telemetry and processor uploads and collect telemetry. Utilized by 1SOP for LADO system. See Figure 5 for an example of a ground antenna.	SV must interface with AFSCN ground antennas (6)
US Naval Observatory	Passive	Provides UTC time for the GPS timing services	System delivers UTC timing information accurate to USNO clocks (6)
End User	Active	Receives PVT signal through GPS receiver	Military and Civil uses expect accurate positioning, timing and velocity information on demand. Though there are some candidate backup systems exist for (VOR/DME, Loran-C, Galileo), no centralized implementation plan exists in case GPS were to go down. The impact of losing PNT beyond private users is outlined below. According to DoT study (7): - *Aviation: loss of IFR, ADS-B, **trouble with precision approaches** - *Maritime: **Collisions in restricted channels during bad weather**

			- Railroads: Cannot track rail anomalies, train operations degraded - *HAZMAT/Emergency services: **Delayed response time,** loss of communication links DHS study (8) also identified: - Communications sector will have loss of cell phone services among other effects - Energy sector depends on GPS for power grid reliability/efficiency, synchronizing services among power networks and locating malfunctions within transmission networks. Oil and gas drilling uses GPS for location/orientation DOD impacts more elusive “There's not a military mission that doesn't depend on space. “ – Heather Wilson SECAF (9) - No doubt affects movement of assets and precision of weapon systems Another area affected is commerce (12): -Transactions for credit cards/ATMs rely on timing information. -NYSE uses timing information for exchange operations. Bolded items above represent critical safety concerns identified by (7) that are caused by a GPS outage
Air Force Monitor Stations	Active	6 USAF monitor stations track GPS Satellites and receive PVT signal	SV must interface with AF Monitor stations (6)
USAF Dedicated Ground Antennas	Active	4 ground antennas dedicated to GPS upload commands, telemetry and processor uploads and collect telemetry.	SV must interface with ground antennas (6)

National Telecommunications and Information Administration	Passive	US Agency that provides limitations on frequencies available for radio communication. Regulates government operated satellites	SV frequencies must not break any communication regulations established by NTIA (10)
Other SVs and their stakeholders	Passive	SVs flying in the same space as the system	SV/launch system must neither become uncontrolled space debris nor collide with any other SV. GPS communications must not have unintended interface with those of other spacecraft or users (10)
Navigation Information Service (USCG)	Passive	Distributes Decoded GPS Data to Civilian Users	Receive accurate and timely GPS status updates to provide to the public. (11)

Figure 5: AFSCN Antenna at Thule Air Base, Greenland (26)

Figure 6: Space and Missile Systems Center, Los Angeles AFB, CA (27)

4.0 Concept of Operations

4.1 Executive Summary

The United States today has no backup to the constellation of GPS satellites on orbit that provide global position, navigation and timing (PNT) information. In the event of a global outage, most enterprises will be adversely affected. However, the most critical areas that will require PNT information are the aviation industry, emergency services, defense and to a lesser extent maritime operations. The system of interest for this paper will be a rapidly deployable small satellite that can be put on orbit to fill in gaps of PNT coverage using existing infrastructure. This system will depart from legacy systems by trading long term performance capability for a simpler system with the ability to quickly be acquired, deployed and operated to sustain critical infrastructure until a more permanent solution can be implemented.

4.2 Need Statement

The US has a need to rapidly acquire and operate a system capable of providing PVT information to military and civil users in case a gap in PNT coverages develops over the Continental United States.

4.3 Enterprises Supported

- IFR Flight Operations
- Emergency Services
- Defense
- Maritime
- Power
- Financial Sector
- Rapid Launch
- Satellite Positioning

4.4 Drivers and Constraints

- Communication and orbital debris regulations
- Cost and schedule for the launch vehicle will be the major driver for this system. Launching one satellite at a time from a Falcon 9/Delta IV/Atlas V is not practical from a cost or schedule perspective to achieve system goals (very expensive, ~6 months of prep time).
- A smaller launch vehicle that can be quickly called up and deployed may be able to support the system objectives. A smaller launch vehicle will also require smaller satellite mass. Conversely if

the entire constellation is small enough to be put on an existing LV with preexisting interfaces (ie. Cubesat deployers), then the mission can realistically happen.

- If operations are in LEO, satellite life may be reduced due to atmospheric drag and delta-V budget available for station keeping. Shorter lifetime may result.
- Technology to build and implement the system should already be available or currently in a “near horizon” state.
- System operation is constrained to existing GPS control segment architecture.
- Engineering tradeoffs to achieve capability will ultimately result in shorter mission duration and lower performance capability than traditional GPS satellites
- System may be susceptible to same risks that affected original GPS constellation to cause the PNT outage.

4.5 Operational Description

Storage – Since time and severity of a loss in PVT coverage is unknown, this system will be most effective if it is already manufactured and ready to go. Therefore, long term SV storage facilities need to be incorporated in the system lifecycle sustainment plan. Where it is stored should depend on the launch vehicle for efficient launch integration and deployment. If the system is launched from a rocket or deployed from an aircraft or space plane, a new facility may have to be constructed near the launch site. Another option is to repurpose existing ICBM infrastructure to have selected launch vehicles fitted with GPS payload so they could be ready to go in case a need develops. This storage method has its benefits as storage infrastructure is designed to protect spaceflight capable hardware against nuclear attacks and maintenance could just be allocated from already funded inline work. Justification for this option can come from comparing the risk of a nuclear attack vs. loss of GPS constellation or political pressure to reduce nuclear stockpiles. One final *avantgarde* option exists. According to (13), the paper suggests a method where a single launch vehicle is selected for constellation deployment using Lagrange point 1. In this method. the entire constellation is flown out to L1 on a single rocket and then launched back at the Earth as it rotates. This method could be viable if global coverage is required quickly. However, development would have to be done on the L1/Earth return system. If this method were used, it may be practical to store the entire constellation out in L1, or even closer to the Moon…

Launch – When a PNT coverage gap develops, the system would have to be removed from storage and integrated onto a launch vehicle for orbital insertion. The most traditional method is using a large rocket like the Falcon 9 depicted in Figure 7, however alternatives for smaller, more rapidly deployable launch systems exist. Existing small rockets that could be used for launch include the Electron for a surface launch or an air launch system like the Pegasus. Though Pegasus is not currently flown, aircraft emerging capable of providing air based space launch capability are Virgin Orbit's Cosmic Girl and StratoLaunch's StratoLaunch aircraft. If ICBM infrastructure is reallocated, the integration work would already have been done and the system can just be launched on demand. If the system can be integrated onto a standard Cubesat deployer, spots on existing launches with Cubesat deployers could be "hijacked" in the name of preserving life and limb. This method will be highly reliant on what vehicle is ready to launch, but hosted loads are currently the most utilized deployment system for small satellites. Another deployment option exists to use a reusable lifting body (aka "space plane") for insertion ops. It would likely have enough delta-V to set up satellites in a LEO single plane, land, refill and continue to set up the constellation. From a space segment perspective, launch and early on orbit checkout would be performed by the 1st Space Operations Squadron from Schriever AFB, CO using existing infrastructure.

Operation and Sustainment – Regular operation of the system would be performed by the 2nd Space Operation Squadron at Schriever AFB using existing infrastructure. Operation would treat the system as another GPS satellite broadcasting PNT signals that the operational control segment would have to accommodate. By the nature of the system, the PNT signal would have to interface with critical safety stakeholders. The FAA and aircraft in flight would have the most urgent need to ensure safe aircraft operations in the area of the coverage gap as well as the area of the launch solution. See Figures 8-11 and Tables 2-3 for a visual description of the proposed system's operations.

Figure 7: GPS III SV01 Launches on the Space X Falcon 9 (29)

4.6 Operational Context

Figure 8: GPS Space Segment Context Diagram (38)

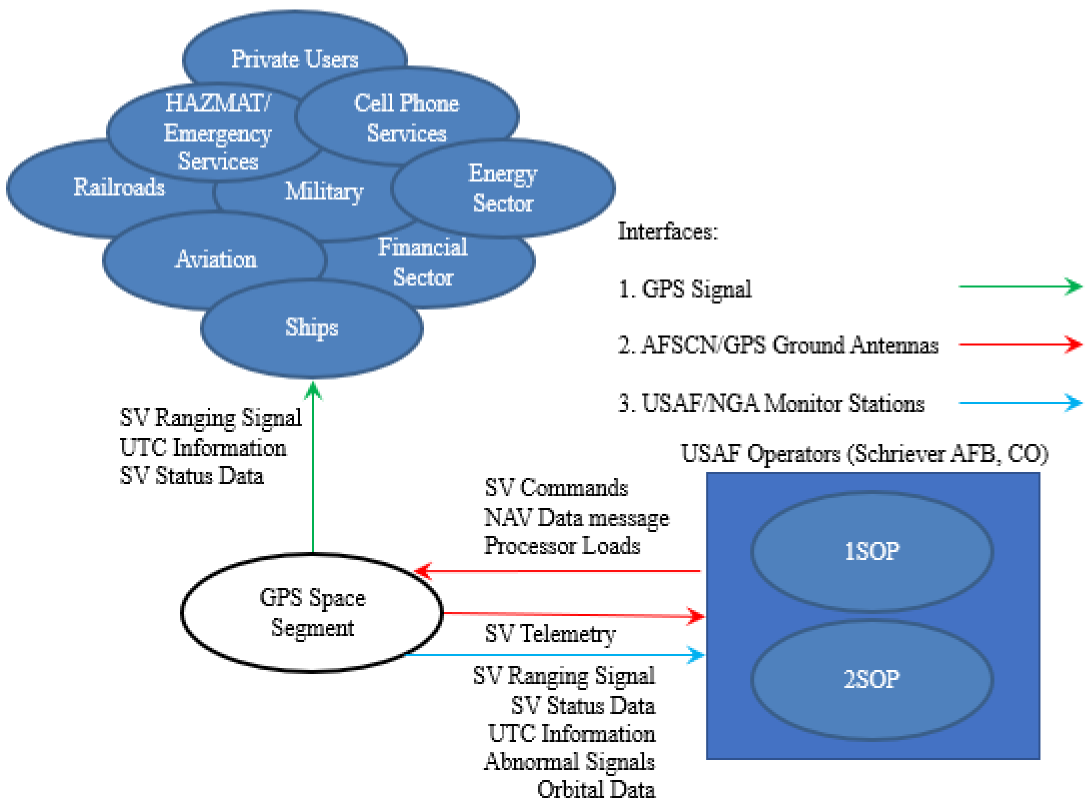


Table 2: Inputs and Outputs to GPS Space Segment (38)

Item	Description
SV Ranging Signal	Signal required by GPS receivers to determine timing, positioning and velocity (must receive signal from 4 separate SVs with adequate geospatial separation)
UTC Information	Coordinated Universal Time – The timing standard generated by the US Naval Observatory that GPS atomic clocks are synchronized to.
SV Commands	Commands sent by operators to control SVs on orbit
NAV Data Message	Updated information used to calibrate the GPS signal
Processor Loads	Updates to the SV software
Raw Pseudo Range/Carrier Phase (SV Ranging Data)	Technical metrics used to determine accuracy of GPS signal
SV Status Data	Data regarding the status of the GPS space vehicle
Orbital Data	Data regarding the orbital parameters of the space vehicle
Abnormal Signals	Operators must be able to receive abnormal signals from SVs in order to troubleshoot and correct problems

Table 3: Context Diagram Stakeholder Decomposition (38)

Context Diagram Label	Active Stakeholders
End User	Aircraft on IFR flight plans, Ships navigating via GPS, Railroad systems, HAZMAT/Emergency responders, Cell phone providers/users, Energy grids, Military, ATM users, stock traders, private users
Ground Antennas (operator interface)	AFSCN, GPS Dedicates Antennas
Monitor Stations (operator interface, receives GPS Signal)	Air Force Monitor Stations, NGA Monitor Stations
Operators	1st Space Operations Squadron, 2nd Space Operations Squadron (primary operator)

Note: for more information on how the GPS signal works, refer to Chapters 2 and 3 of Understanding GPS Principles and Applications 2nd edition (38)

Figure 9: OV-1

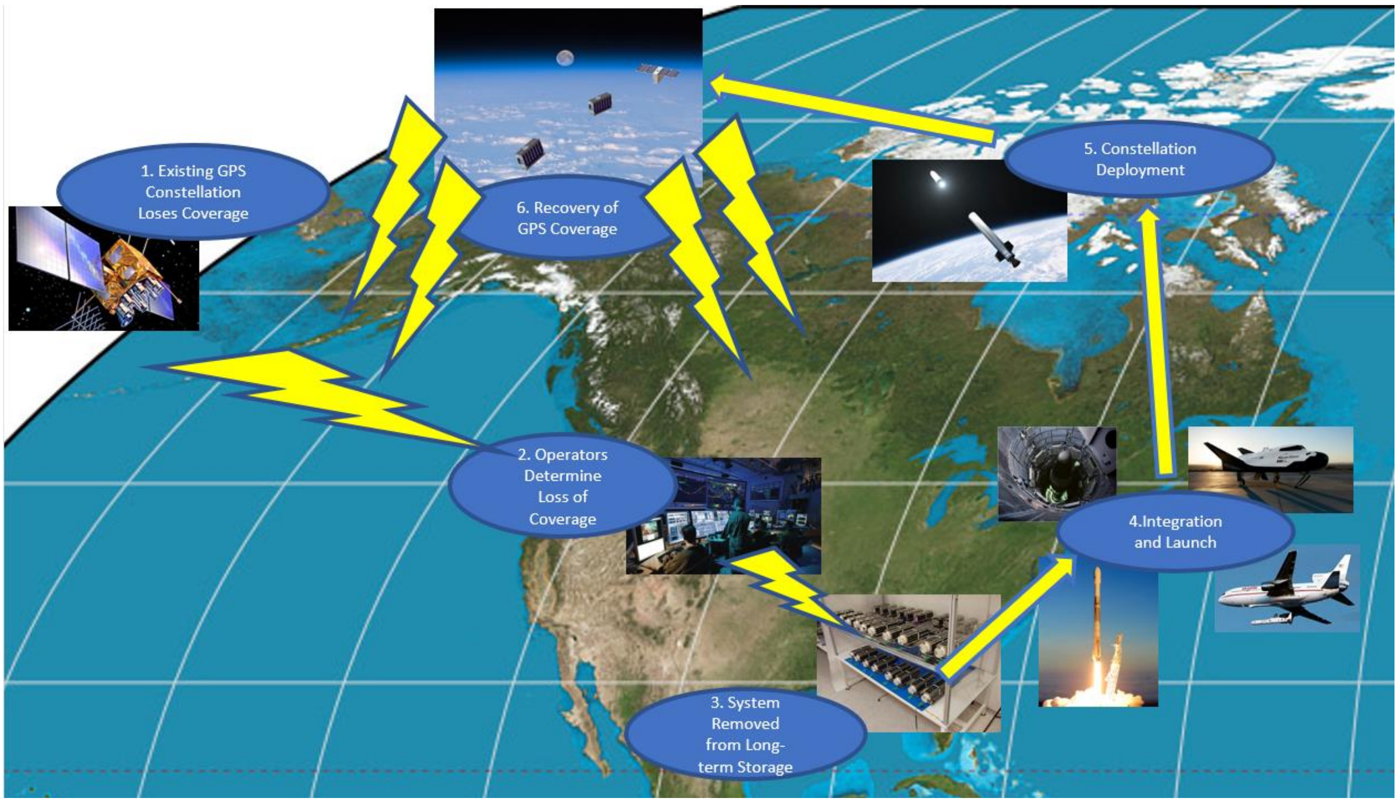


Table 4: OV-1 Description

Stage	Description
1	Existing GPS Constellation Loses Coverage
2	Operators Determine Loss of Coverage
3	System Removed from Long-term Storage
4	Integration and Launch
5	Constellation Deployment
6	Recovery of GPS Coverage

Figure 10: Proposed System Operational Timeline

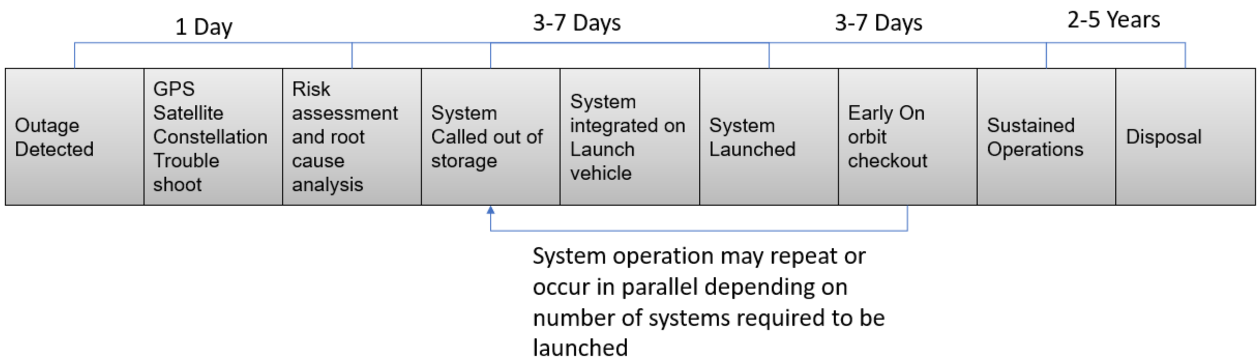
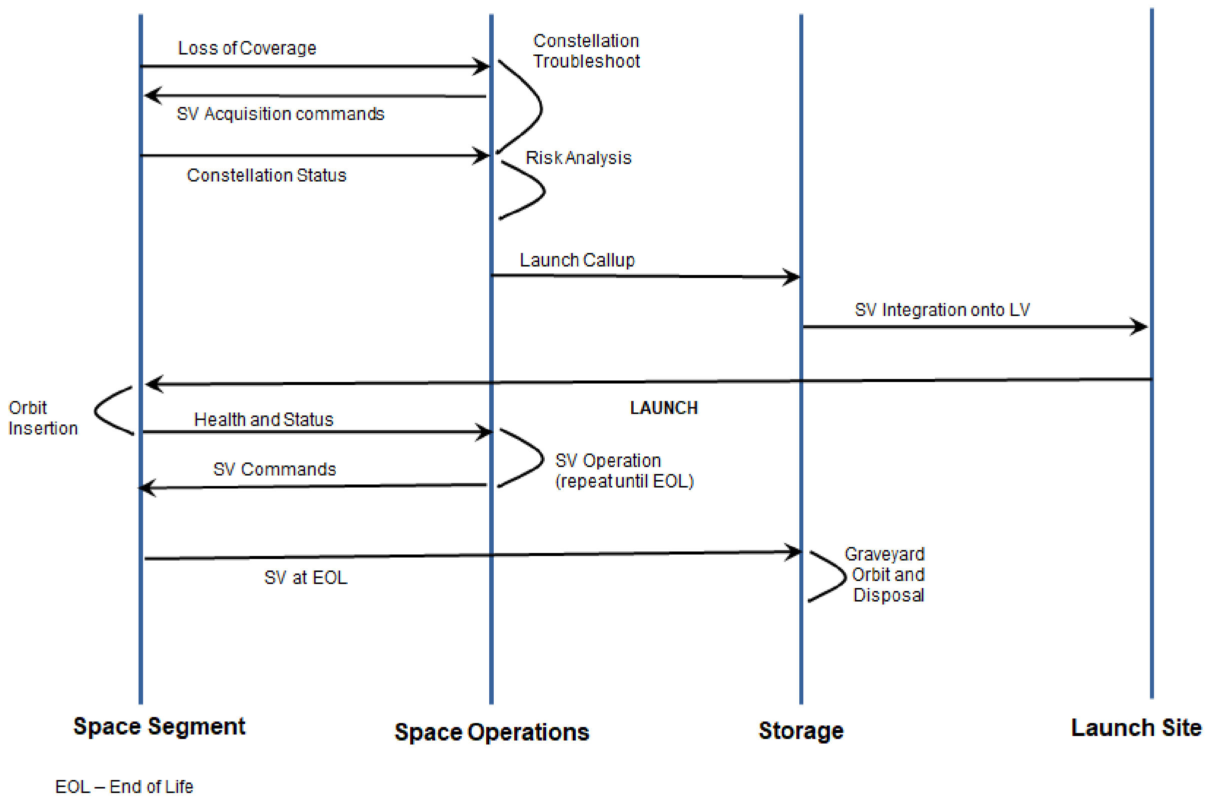


Figure 11: Use Case Description



4.7 Risks

Storage – *If the system is in term storage creates risk, then the system will be unusable after prolonged periods of time.* Storage of satellite batteries is always of concern as they will degrade when they pass their shelf life. Orbital storage has plenty of risks, primarily radiation and inability to remove and replace parts. Ground storage can always be sabotaged if the PVT coverage gap is created maliciously. If the gap is caused by a severe solar storm, it is possible the solar storm could affect the satellites in storage as well.

Launch – *If the launch vehicle relies on GPS navigation, then the system would fail if the launch site does not have proper coverage and the mission would be ironically cut short.* Launch vehicles use GPS as well, so this specific case must be seriously considered.

Operations – If the root cause of the GPS coverage outage is not mitigated, then the system deployed would also be at risk of losing coverage. If the outage occurs during a solar storm and the system is launched before the storm has ended, the system has a risk of being affected as well. If the source is malicious, military operations should end the threat so that it does not take out the contingency system. Standard space environmental risks apply here as well and what environmental risks apply can depend on system flight altitude.

4.8 Organizational Impact

Programmatic – At the very least a new three letter sub directorate of the Global Positioning directorate (2 letter – “GP”) would have to be set up at SMC. For example, GPV is GP Space Vehicle, GPG is GP Ground ect…

Operational – 1SOP/2SOP Space operators would need to be trained in deployment and operation of the new vehicles while operating existing satellites.

Launch and Integration – Launch crews would need to be able to integrate and launch the new system efficiently and safely. Launching any system is not a quick and easy task so extra attention would be made here to ensure successful mission ops.

5.0 System Requirements

Table 5: Functional and Non-functional Requirements

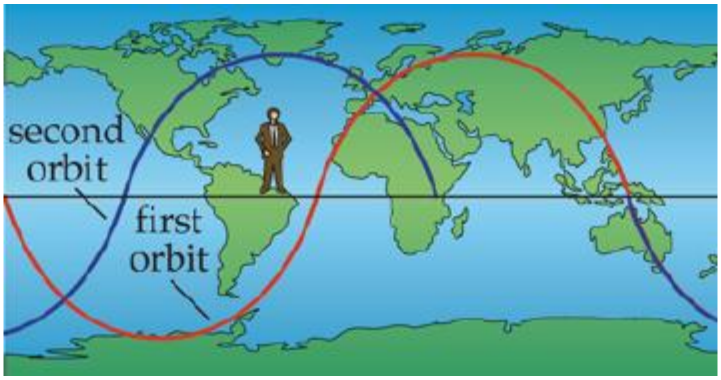
Type	Requirement	Rationale
Functional	The system shall fill coverage gaps within the existing GPS Constellation when they occur	No back-up to the current GPS system exists
	The system shall sustain response time for HAZMAT and emergency services	Delayed response times will cause loss of life
	The system shall allow unrestricted maritime navigation of US waterways	Safety concern with ships operating in constricted waterways with restricted visibility
	The system shall sustain operations of power grid	The US power grid relies on GPS to synchronize its 9 power networks
	The system shall sustain US economy's ability to conduct transactions	US economy requires the time of transactions be precise for efficient operations
Non-functional	The system shall provide accuracies better than 13m in the horizontal plane and 22m in the vertical plane.	GPS PNT signal requirements
	The system shall provide UTC time dissemination better than 40 ns	GPS PNT signal requirements
	Position Dillution of Precision shall be <= 6	Aerospace Cooperation metric to indicate GPS coverage based on spacing of PNT signal sources
	The system shall integrate on existing launch systems	The system will deploy faster if the launch system is already available
	The system shall be put on orbit expeditiously in the case that an urgent need arises	GPS outages can cause safety concerns for key infrastructure if it cannot be restored quickly
	The system shall provide GPS PNT data with enough accuracy to allow safe aircraft operation under IFR conditions	Aircraft cannot fly in the clouds without an IFR clearance. Safety concern with takeoff and landings under IFR conditions as well as aircraft low on fuel in the weather. Economic impact will be devastating if Aircraft cannot fly on IFR
	The system shall cover 99% of the Continental United States for 1 hour	Protection of life in the United States in the event of severe GPS outages is essential.
	The system shall fill a hole in GPS coverage within 2 weeks of outage	Response to a rapid deterioration in GPS coverage is essential to preserve life
	The system shall have a 98% availability for single SVs to the operators	Operators must be able to check the health and status and send commands to SVs on orbit when they need to

	The system shall interface with AFSCN antennas, NGA monitor stations and existing GPS monitor stations and antennas.	Use of existing infrastructure will allow for faster/cheaper acquisition and easier network integration
	The system shall be NTIA compliant	Communication regulations required for all US government space vehicles
	The system shall abide by USG orbital debris regulations	Orbital debris is increasingly becoming a hazard for all space users
	The system shall have a minimum mission life of 2 years and de orbit no later than 25 years	The system must last long enough until a more permanent solution can be implemented. Trades longevity for rapid deployment capability. 25 years is a maximum limit for Cubesats in LEO
	A single space vehicle shall be manufacturable within 1 year in the production and deployment phase of acquisition	The system must be acquired quickly since the timing of a severe GPS disruption is unknown

6.0 Orbital Analysis

The first part in technical design work will begin with determining which altitude to fly in. Altitude will drive many key attributes like delta-V for station keeping, power requirements for communications and number of space vehicles required for coverage. Since the GPS constellation has a 12 hour period with ~180 deg node displacement (see Figure 12 for a depiction of Node Displacement) per revolution, a key feature of the existing GPS constellation is that the ground track will repeat about every sidereal day. Therefore, the ground track of the recovery constellation must cover the same spot every day to match the coverage gap (15). If there is a complete outage, a constellation with at quadruple full Earth will have to be established - but for the purpose of this paper the starting point in design will be a 1 hour window where only three GPS satellites are in view of the Continental United States due to the redundancy built into the existing GPS constellation. Since the system will be flying in LEO, step 1 will be to determine which acceptable LEO altitudes will have a 360 deg node shift every sidereal day.

Figure 12: Example of Ground Track Node Displacement (14)



First, number of orbits per sidereal day was determined

$$(1) Period = (2\pi \sqrt{\frac{(6371km + altitude)^3}{3.986e5 km^3 / s^2}}) \frac{1 \min}{60 s}$$

$$(2) Node\_Diplacement = -(15° / hr)(\frac{1hr}{60 \min})P - 360°)$$

$$(3) 360° NodeDisplacments / SRDay = \frac{(1436.068 / P) * N\_D}{360°}$$

$$(4) Orbits / SRDay = 1436.068 / P$$

Using this method, Table 6 and 7 determines a whole number of orbits/day is required for repeat ground track.

Table 6: Altitude Determination

,

Altitude (km)	Period (min)	ND/P (deg)	360 deg ND/SR day	Orbits/SR day	SMAD Altitude (km)
LEO					
200	88.35012567	337.9124686	15.25701417	16.25428361	
270	89.76565231	337.5585869	15.00069819	15.99796763	274.419
300	90.37459868	337.4063503	14.89290347	15.89017291	
400	92.41430281	336.8964243	14.54218664	15.53945608	
500	94.46912517	336.3827187	14.20418391	15.20145336	
560	95.7092293	336.0726927	14.00721874	15.00448818	566.896
600	96.53895533	335.8652612	13.87825925	14.87552869	
700	98.62368524	335.3440787	13.56381694	14.56108638	
800	100.7232092	334.8191977	13.26029852	14.25756796	
890	102.625344	334.343664	12.99603809	13.99330754	893.795
900	102.8374236	334.2906441	12.96717997	13.96444941	

Note: Assumes 0˚ inclination as a starting point

Table 7: SMAD Orbital Dynamics Verification (17)

Return to Navigator		Orbit Dynamics	
(All information on this sheet is contained in the bloc			
Circular orbit altitude		893.795	km
Semi-major axis		7271.932	km
Inclination		99.01	deg
Eccentricity		0.0000	
Perigee altitude		N/A	km
Apogee altitude		N/A	km
Repeating ground tracks			
Number of orbits	*14*		
Number of days Set to value	*1*	*7271.932*	

Though the methodology was verified, the SMAD sheet revealed that orbital inclination was not constrained. Inclination will affect repeat ground tracks due to the J2 effect, where Earth's oblateness will perturb orbits by causing nodal precision. Therefore, either altitude or inclination must be constrained if a stable orbit with a repeat ground track is desired. If we want to cover specific regions, the inclination was be constrained with altitude set as a variable. To cover the entire CONUS region, an inclination of 40˚ was selected to maximize coverage time. New possible altitudes determined from the SMAD Spreadsheet are listed in Table 8.

Table 8: Potential LEO Mission Altitude

Orbits/Sidereal Day	Altitude (km)
16	178.659
15	479.441
14	814.117

Due to small satellite orbital debris regulations, beyond 800 km is not desirable for CubeSats. However, to increase coverage per satellite an altitude of 814.117 km can be used given that the satellite has enough End of Life delta-V to de-orbit within 25 years to abide by orbital debris requirements. As a final sanity check, the altitude of the GPS constellation was determined analytically from the SMAD spread sheet in Table 9. According to gps.gov (39), the altitude of the GPS constellation is approximately 20,200 km - therefore enough confidence is provided to validate the methodology and continue onward.

Table 9: Altitude of the GPS Constellation (17)

Return to Navigator		Orbit Dynamics	
		(All information on this sheet is contained in the block	
Circular orbit altitude		20181.656	km
Semi-major axis		26559.793	km
Inclination	*55.00*	55.00	deg
Eccentricity		0.0000	
Perigee altitude		N/A	km
Apogee altitude		N/A	km
Repeating ground tracks			
Number of orbits	*2*		
Number of days / Set to value	*1*	*26559.793*	

According to the SaVi (32) results in Figures 13-14 and Table 10, it will take 5 satellites flying at an altitude of 814 km to provide 99% CONUS coverage for roughly one hour. According to the simulation, some parts of the very southern Texas border get momentarily clipped however some minor constellation phasing adjustments can trade 4.5 minutes of coverage for that meet objective requirements. This trade is worthwhile to national security interests because the US Air Force conducts pilot training at Laughlin Air Force base and the Border Patrol conducts security operations in that area. The results use a transmitter strong enough that the half-angle beam width covers the entire swath width of the space vehicle, enabling the PNT signal to be distributed effectively to the entire section of Earth which is physically in line of view of the Space Vehicle.

Figure 13: Full CONUS Coverage, T= 00:00:00 (32)

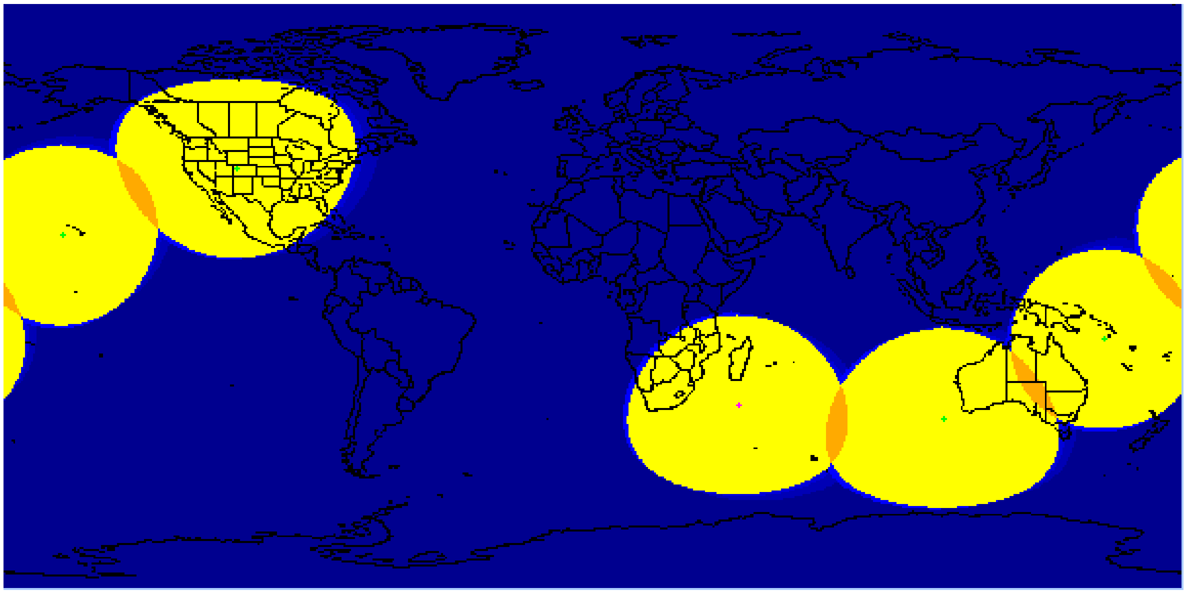


Figure 14: Full CONUS Coverage, T= 01:04:30 (32)

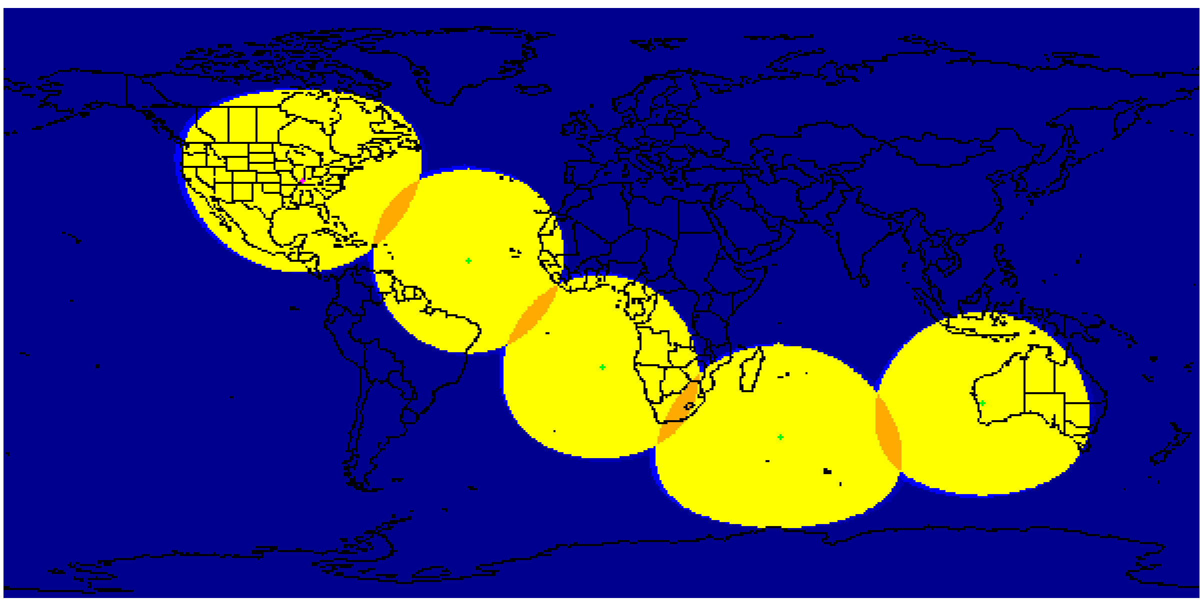


Table 10: SaVi Input Constellation Orbital Parameters (32)

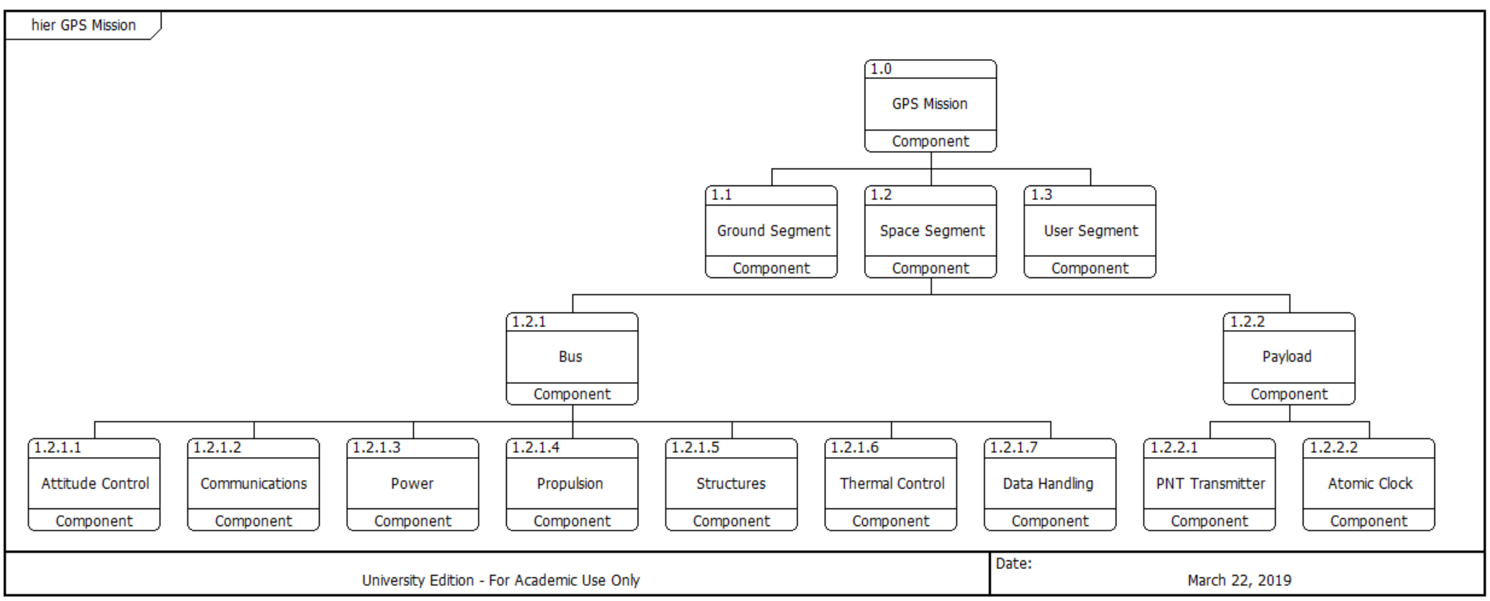
no.	semi-major axis	eccentricity	inclination	longitude asc. node	arg. periapsis	time to periapsis	satellite name
1	7192.14	0.0000	40.000	0.000	0.000	0.000	SV01
2	7192.14	0.0000	40.000	0.000	0.000	850.000	SV02
3	7192.14	0.0000	40.000	0.000	0.000	1700.000	SV03
4	7192.14	0.0000	40.000	0.000	0.000	2550.000	SV04
5	7192.14	0.0000	40.000	0.000	0.000	3400.000	SV05

7.0 SV Architecture Analysis

7.1 Physical Architecture Hierarchy

Now that the mission orbit is worked out, some preliminary design work with the space system's physical architecture (see Figure 15) can begin. The architecture analysis began with the mission payload. GPS satellites currently utilize Rubidium Atomic Frequency Standards (RAFS) manufactured by Excelitas. Currently RAFS are the primary clock used in GPS because they are the smallest and lightest unit capable of providing the clock stability and drift rate required to complete the GPS mission. As a rule of thumb, 1 nanosecond leads to 1 ft of accuracy in position determination. Therefore, the RAFS will be the starting point for the payload analysis. Technical information from Excelitas on the RAFS is listed in Figure 16.

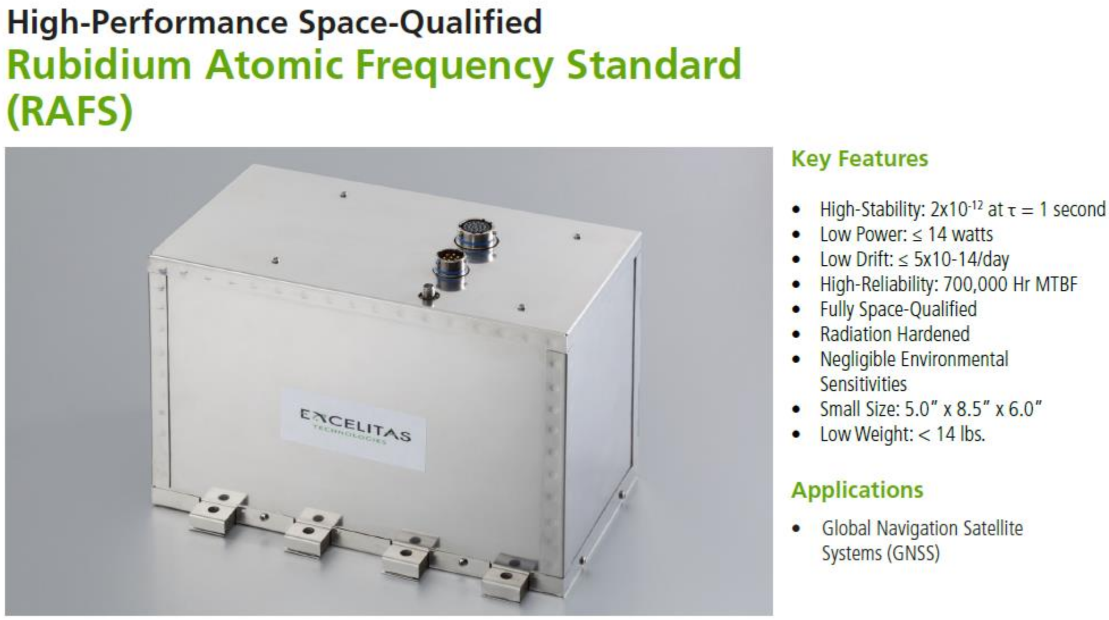
Figure 15: Top Level SV Physical Architecture (40)



Note: Data handling and power for the Payload provided by the Bus architecture in this study

7.2 Payload Analysis and Preliminary SV Sizing

Figure 16: Excelitas RAFS Datasheet (30)



After the frequency standard for the system was selected, the next step of the payload analysis is to determine how much power is required to get the signal to Earth. The approach taken was to determine how much signal power is received by GPS receiver on the ground, and then design a link budget that will match the power flux of the original GPS signal. Research into the existing GPS link budget was conducted and a document from the FCC that does the calculations is presented in the first half of Figure 17 (31). Since the estimated received power is -158.43 dBW, the system in this research will be designed to put a -160 dBW signal on the Earth's surface under worst case conditions.

Figure 17: GPS L1 Link Budget (31)

GPS L1 Link Budget

Satellite Tranmitter

Transmitter Power (25 Watts)	14.25 dBW	
RF Losses in trasmitter path	-1.25 dB	
Antenna Gain (with respect to an isotrope)	13.5 dBi	
Satellite EIRP (wrt isotropic radiator)	**26.50 dBW**	**446.68 Watts**

Propagation

Atmoshperic and Polarization Losses	-0.5 dB

$$Free\ Space\ Path\ Loss = -10\times\log_{10}\left[\left(\frac{4\pi d}{\lambda}\right)^2\right]$$

where d = distance from antenna =	2.52E+07 meters
c = speed of light =	3.00E+08 m/sec
f = frequency =	1.58E+09 Hz
lambda = wavelength = c/f =	1.90E-01 meters

$$= -10\log_{10}\left[3.17\text{E+08} / 1.90\text{E-01}\right]^2$$

$$= -10\log_{10}\left[1.67\text{E+09}\right]^2$$

Free Space Path Loss over Distance	-184.43 dB

Received Power on Earth	**-158.43 dBW**	
	-128.43 dBm	**1.44E-04 pW**

Calculated EIRP required for a Rapidly Deployable GPS System at 814.117 km altitude:

L1 Signal Characteristics:
Frequency: 1.575 Ghz
Data rate: 50 bps
Eb/No: 9.60 dB (BPSK)
Bit Error: 1.0e-5

814.117 km/0 deg elevation angle, nadir angle = 62.47˚
Required signal Beamwidth = 125.12˚

Worst Case Loss @ elevation angle = 90˚
Space loss = -172.49 dB
Atmosphere Attenuation = -.036

EIRP – Space Loss – Atmosphere Loss = Received EIRP
EIRP = -160 + 172.49 + .036
EIRP = 12.526 dB

These calculations were used to find transmitter mass and power requirements (Tables 11-14). Project will assume use of the GPS L1 signal since it is the most commonly relied upon out of all the GPS signals. Figure 18 depicts the geometry involved in determining satellite coverage.

Figure 18: Satellite Coverage Geometry (33)

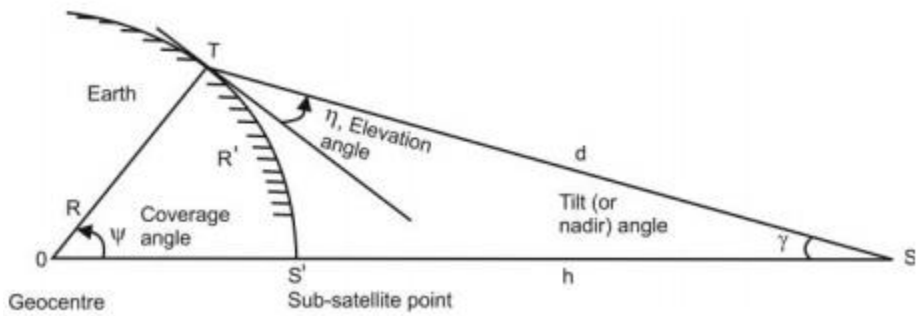


Table 11: Payload Transmitter Sizing (17)

Return to Navigator

Communications System - Downlink

(All information on this sheet is contained in the block from Cell A1 to Cell Q33)

Frequency	1.58	1.575	GHz
Wavelength		1.90E-01	m
Spacecraft Transmitter			
Output power		20.92	W
Output power		13.21	dB
Line loss		-3.00	dB
Antenna efficiency		55.0%	
Antenna diameter		0.11	m
Peak antenna gain		2.32	dB
Half-power beamwidth	124.94	124.94	deg
EIRP	12.53	12.53	dB
Pointing error			deg
Antenna pointing loss			dB
Duty cycle (per orbit period)		100.0%	

Data rate	5.000E+01	5.000E+01	bps
Geometry & Atmosphere			
Altitude		814.117	km
Planet angular radius		62.47	deg
Elevation angle	90.00	90.00	deg
Nadir angle		0.00	deg
Planet central angle		0.00	deg
Propagation path length		-6378.137	km
Atmospheric attenuation at zenith		-0.060	dB
Rain attenuation		0.000	dB
Increase in system noise temp		0.00	K
Link Budget			
EIRP		12.53	dB
Space loss		-172.49	dB
Atmospheric attenuation		-0.36	dB
Rain attenuation		0.00	dB
G/T			dB
Antenna pointing losses			dB
Eb/No			dB
C/No			dB
Implementation loss		-2.00	dB
Margin			dB

Probability of Bit Error		1.00E-05	
Required Eb/No		9.60	dB
Ground Receiver			
Antenna efficiency		55.0%	
Antenna diameter			m
Peak antenna gain			dB
Half-power beamwidth			deg
Pointing error			deg
Antenna pointing loss			dB
System noise temperature		260.00	K
G/T			dB/K
Mass & Power Estimates			
Transmit Antenna			
Mass		0.1	kg
	TWTA	SSPA	
Transmitter mass	4.3	1.1	kg
Peak transmitter input power	42.1	73.7	W
Average transmitter input power	42.1	73.7	W

Table 12: Transmitter Selection (17)

Transmitter Type	Payload Mass (RAFS+Transmitter+Antenna) kg	Payload Power (RAFS peak+transmitter peak) W	Preliminary Loaded Mass (kg)	Preliminary Power Required (W)
Traveling Wave Tube (TWTA)	6.3+4.3+.1 = 10.7	65+42.1= 107.1	56.6	343.1
Solid State Power Amplifier (SSPA)	6.3+1.1+.1= 7.5	65+73.7=138.7	39.3	420.0

Table 13: Preliminary Spacecraft Sizing (TWTA) (17)

Return to Navigator

Preliminary Spacecraft Sizing

(All information on this sheet is contained in the block from Cell A1 to Cell AG24)

					Mass (kg)	Average Power (W)
Payload Mass	*10.7*	10.7	kg	Payload	10.7	107.1
Payload Percentage		27.0%		S/C Subsystems	29.1	167.4
Margin Percentage (Mass)		25.0%		*ADCS*	*3.2*	*32.5*
				C&DH	*1.7*	*13.5*
Payload Power (peak)	*107.1*	107.1	W	*Power*	*11.2*	*60.6*
Payload Percentage		39.0%		*Propulsion*	*1.5*	*10.2*
Margin Percentage (Power)		25.0%		*Structure*	*8.7*	*0.0*
				Thermal	*1.4*	*10.2*
Required Total Delta-V		250.2	m/s	*TT&C (Communications)*	*1.4*	*40.4*
Delta-V Margin Percentage		20.0%		Margin	10.0	68.6
Propellant Specific Impulse	*240.0*	240.0	sec	S/C Dry Mass	49.8	
				Propellant Mass	6.8	
				S/C Loaded Mass	56.6	
				S/C Power		343.1

Table 14: Preliminary Spacecraft Sizing (SSPA) (17)

Return to Navigator

Preliminary Spacecraft Sizing

(All information on this sheet is contained in the block from Cell A1 to Cell AG24)

					Mass (kg)	Average Power (W)
Payload Mass	*7.5*	7.5	kg	Payload	7.5	138.7
Payload Percentage		27.0%		S/C Subsystems	20.2	197.3
Margin Percentage (Mass)		25.0%		*ADCS*	*2.2*	*38.5*
				C&DH	*1.2*	*15.9*
Payload Power (peak)	*138.7*	138.7	W	*Power*	*7.8*	*71.1*
Payload Percentage		41.3%		*Propulsion*	*1.0*	*12.0*
Margin Percentage (Power)		25.0%		*Structure*	*6.1*	*0.0*
				Thermal	*1.0*	*12.0*
Required Total Delta-V		250.2	m/s	*TT&C (Communications)*	*0.9*	*47.8*
Delta-V Margin Percentage		20.0%		Margin	6.9	84.0
Propellant Specific Impulse	*240.0*	240.0	sec	S/C Dry Mass	34.6	
				Propellant Mass	4.7	
				S/C Loaded Mass	39.3	
				S/C Power		420.0

Table 15: Cubesat Size Specifications (18)

PAYLOAD SPECIFICATION FOR 3U, 6U, 12U AND 27U

3. PARAMETERS

Symbol	Parameter	Conditions	Unit	3U		6U		12U		27U	
				Min	Max	Min	Max	Min	Max	Min	Max
M	Mass	At launch	kg [lb]	-	6.0 [13.2]	-	12.0 [26.4]	-	24.0 [52.9]	-	54.0 [119.0]
CM_X	Center of mass, X	Stowed in CSD	mm [in]	-20 [-.79]	20 [.79]	-40 [-1.57]	40 [1.57]	-40 [-1.57]	40 [1.57]	-60 [-2.36]	60 [2.36]
CM_Y	Center of mass, Y	Stowed in CSD	mm [in]	10 [.39]	70 [2.76]	10 [.39]	70 [2.76]	55 [2.17]	125 [4.92]	100 [3.94]	180 [7.09]
CM_Z	Center of mass, Z	Stowed in CSD	mm [in]	133 [5.24]	233 [9.17]	133 [5.24]	233 [9.17]	133 [5.24]	233 [9.17]	133 [5.24]	233 [9.17]
Height	Maximum payload depth, +Y dimension		mm [in]	-	109.7 [4.319]	-	109.7 [4.319]	-	222.8 [8.771]	-	332.8 [13.102]
Width	Maximum payload width from origin, ±X dimension		mm [in]	-	56.55 [2.226]	-	119.7 [4.713]	-	119.7 [4.713]	-	176.25 [6.939]
Tab Width	±X dimension		mm [in]	112.7 [4.437]	113.1 [4.453]	239.0 [9.409]	239.4 [9.425]	239.0 [9.409]	239.4 [9.425]	352.1 [13.862]	352.5 [13.878]
Tab Length	+Z dimension		mm [in]	361 [14.21]	366 [14.41]	361 [14.21]	366 [14.41]	361 [14.21]	366 [14.41]	361 [14.21]	366 [14.41]
EP_Y	Ejection plate contact zone, +Y dimension from origin		mm [in]	-	100 [3.94]	-	100 [3.94]	-	213 [8.39]	-	326 [12.84]
DC_X1	Deployable contact zone with CSD, ±X face near +Y face		mm [in]	91.4 [3.598]	-	91.4 [3.598]	-	204.5 [8.051]	-	317.6 [12.504]	-
DC_X2	Deployable contact zone with CSD, ±X face near -Y face		mm [in]	-	20.3 [.799]	-	20.3 [.799]	-	20.3 [.799]	-	20.3 [.799]
DC_+Y	Deployable contact zone with CSD, +Y face (1)		mm [in]	43.85 [1.726]	-	107.0 [4.213]	-	107.0 [4.213]	-	163.55 [6.439]	-
DC_-Y	Deployable contact zone with CSD, -Y face (1)		mm [in]	31.2 [1.228]	-	94.3 [3.713]	-	94.3 [3.713]	-	150.9 [5.941]	-
F_{DS}	Force from optional deployment switches, summated, Z axis (2)	When contacting CSD ejection plate. Per CSD ejection Spring.	N	-	5.0	-	5.0	-	5.0	-	5.0
D_{DS}	Payload separation from ejection plate necessary to change deployment switch state, Z axis	Is switches reside on –Z face.	mm [in]	1.3 [.05]	12.7 [.50]	1.3 [.05]	12.7 [.50]	1.3 [.05]	12.7 [.50]	1.3 [.05]	12.7 [.50]
F_{FD}	Friction force deployables impart on CSD walls during ejection	summated (all 4 sides), per CSD ejection spring	N	-	2.0	-	2.0	-	2.0	-	2.0
TML	Total Mass Loss	Per ASTM E 595-77/84/90	%	-	1.0	-	1.0	-	1.0	-	1.0
CVCM	Collected Volatile Condensable Material	Per ASTM E 595-77/84/90	%	-	.1	-	.1	-	.1	-	.1
DP	CSD de-pressurization rate	During launch	psi/ sec	-	1.0	-	1.0	-	1.0	-	1.0
D_X	Location of optional separation electrical connector, +X dimension		mm [in]	40.79 [1.601]	41.07 [1.621]	103.95 [4.088]	104.23 [4.108]	103.95 [4.088]	104.23 [4.108]	160.49 [6.314]	160.77 [6.334]

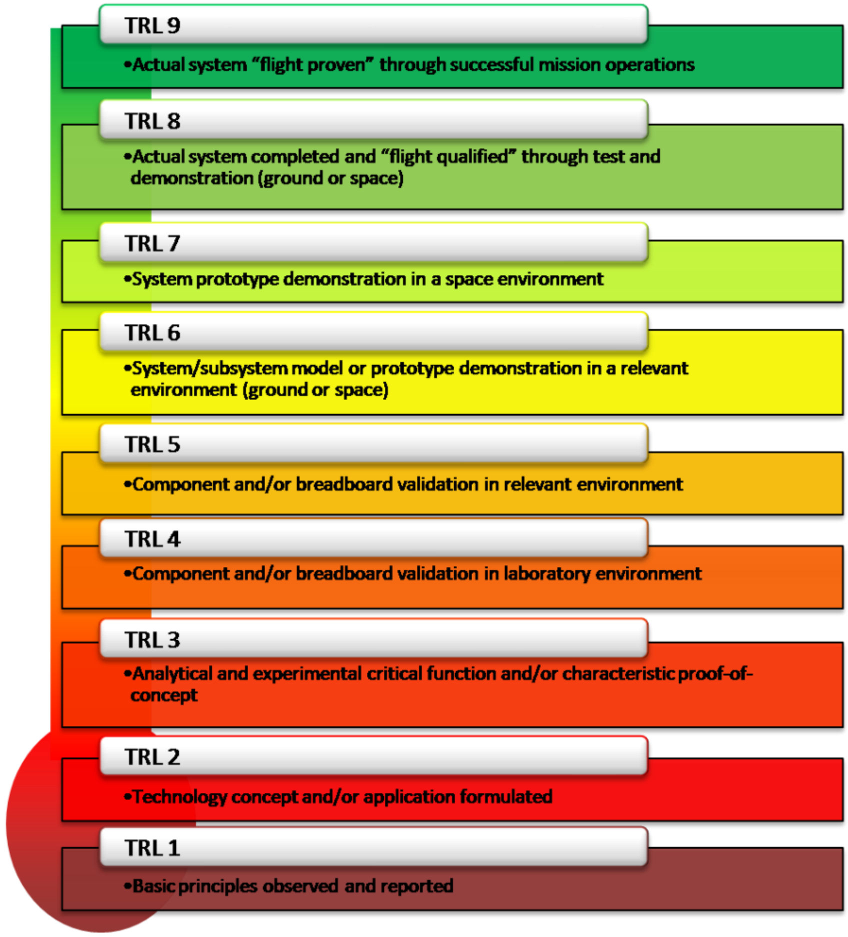
(1) Some contact zones are not present on the 3U. Refer to Figure 6-2 for locations.
(2) Ensures payload will not gap from CSD ejection plate prior to separating.

After payload sizing, it was determining the mass of the spacecraft will be in the mass range of a 27U sized Cubesat from the size specification chart in Table 15. Using a Cubesat is ideal for this application because deploying multiple CubeSats on a single launch vehicle is a widely utilized practice which this system can make use of to allow for rapid deployment of a constellation. Since the mass limit is 54 kg according to the high-lighted specification for a 27U Cubesat, the SSPA transmitter was selected since a lighter payload will result in a lighter spacecraft and more margin to potentially add additional features such as more propellant or an extra RAFS.

7.3 SV Subsystem Analysis and TRL Assessment

With the payload and preliminary sizing complete, the next step is to evaluate the different subsystems to assess the technological readiness level (TRL) of the components and create a preliminary cost estimate. TRL is broken up into nine stages: stage one involves basic laboratory observations and stage nine involves technology validation during real life mission operations (see Table 16). In the DoD, a TRL of 6 is required to create a program of record, however in the DoD acquisitions framework there is a Technology Maturation and Risk Reduction phase designed to improve the TRL of sub level 6 technologies (4). Though NASA TRL criteria is used, DoD criteria mimics NASA's close enough to be useful to this project's military stakeholders. Using the SMAD spreadsheet (17) and the 2018 Nasa State of the Art of Small Spacecraft Technology (20), each subsystem in the bus physical architecture will be evaluated at a top level for technological system feasibility and to determine cost estimates. After initial TRL levels for each sub system were determined, equivalent values were inserted into the SMAD cost estimating section to determine the total cost of the constellation. None of the examples used in this paper imply an endorsement.

Table 16: NASA TRL Criteria (34)



7.3.1 Attitude Control

Attitude control is relatively easy to analyze and will not be a problem with the spacecraft. According to (20), all type of components for attitude control are at a TRL level 9 and consumer off the shelf products and be bought and integrated with ease. According to the Table 17, required angular momentum for the system is 0.05 Nms and the maximum according to (20) is 8 Nms of torque storage allowing for plenty of margin. Magnetorquers can also be included in the system for momentum dumping operations but aren’t included in the top-level analysis. An example reaction wheel that gives 10x margin is the RWP500 by Blue Canyon Technologies (21) shown below. Its mass properties were inserted into the SMAD spread sheet for the final sizing analysis.

Table 17: System Reaction Wheel Analysis (21) (17)

RWP500

- Momentum: 0.50 Nms
- Mass: 0.75 kg
- Volume: 11 x 11 x 3.8 cm

Required angular momentum			0.05	N-m-s
Wheel mass			0.75	kg

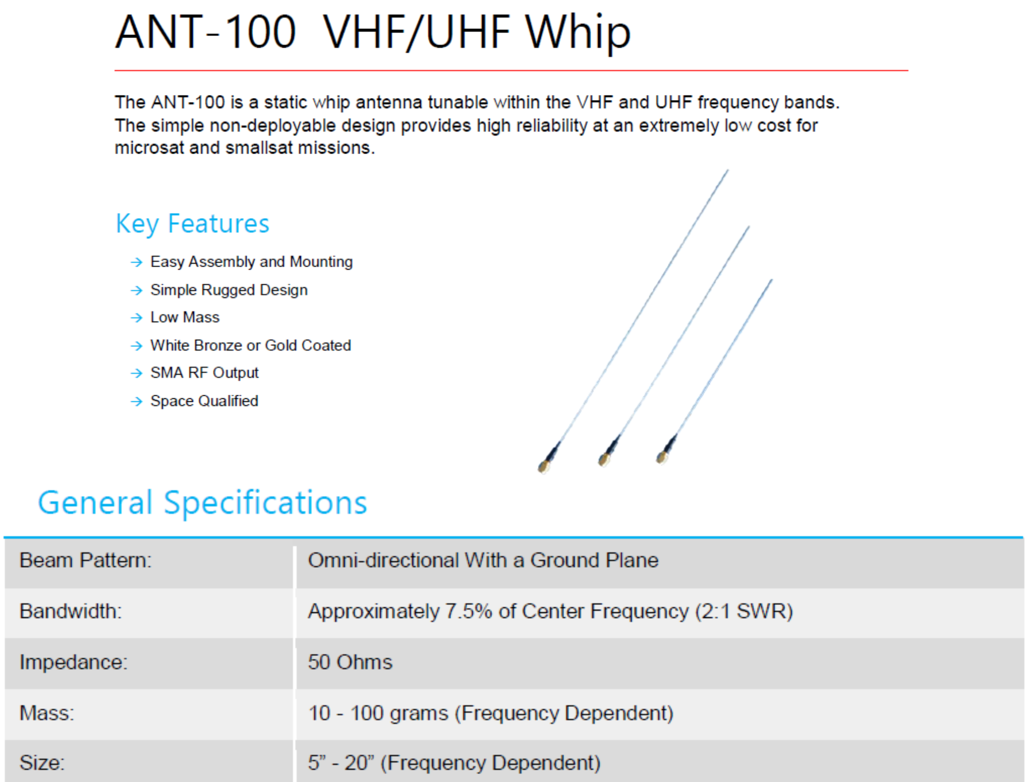
Zero-Momentum System (Reaction Wheel)							
Torque for sizing		3.03E-05	N-m	Wheel radius	*0.06*	0.06	m
Total time sizing torque is applied		1517.6	sec	Wheel angular velocity	*325.0*	325.0	rpm

7.3.2 Communications

The next system is communications. The key aspect of this system is that a separate antenna must be utilized to transmit and receive data since AFSCN uses different uplink/downlink frequencies (17). The uplink/downlink antenna will also be separate from the payload antenna due to the different channel, but the payload antenna's mass is already accounted for in the payload sizing. A list of relevant frequencies is listed below. For the uplink and downlink, SMAD estimates a .1 kg antenna with a 1.1 kg SSPA transmitter on the downlink. As with attitude control, communication is a technology area with TRLS of 9 being very common so communications will not hold back the satellite (20). An example space rated communications antenna by SpaceQuest (22) and relevant communication frequencies is shown in Table 18.

Table 18: Relevant Communication Frequencies (17) (22)

	Command (Uplink)			Telemetry (Downlink)		
Network	Mhz	Band	Bps	Mhz	Band	Bps
AFSCN	1760-1840	L-Band	1-2 k	2200-2300	S-Band	1.25-1.024 M
L1/L1C				1575.42	L-Band	50
L2/L2C				1227.60	L-Band	50
L5				1176.45	L-Band	50



ANT-100 VHF/UHF Whip

The ANT-100 is a static whip antenna tunable within the VHF and UHF frequency bands. The simple non-deployable design provides high reliability at an extremely low cost for microsat and smallsat missions.

Key Features

- Easy Assembly and Mounting
- Simple Rugged Design
- Low Mass
- White Bronze or Gold Coated
- SMA RF Output
- Space Qualified

General Specifications

Beam Pattern:	Omni-directional With a Ground Plane
Bandwidth:	Approximately 7.5% of Center Frequency (2:1 SWR)
Impedance:	50 Ohms
Mass:	10 - 100 grams (Frequency Dependent)
Size:	5" - 20" (Frequency Dependent)

7.3.3 Power

Power is a critical subsystem because it enables the mission to happen. Without power, the spacecraft will be dead on orbit. Many solar arrays are at a TRL of 9, however one solar array that looked promising is at a 7. The MMA Design's eHaWK (High Watts per Kilogram) with a cell efficiency of 28.3% and solar array power density of 120W/kg looks suited for this type of mission due to its size scalability and power density, so its specs were inserted into the SMAD spreadsheet (Table 19). This solar array is scheduled to launch in 2020 and is expected to reach TRL 9 soon. For batteries, an EaglePicher Space Rechargeable Li-ion Battery with a specific energy density of 153.5 W-hr/kg was chosen (see Table 20). Though the battery has a TRL of 7, Eagle Picher has extensive flight experience on military systems and the TRL is expected to increase in the coming years (20).

Table 19: Solar Array Design (17)

Return to Navigator	Power Subsystem - Solar Array Sizing						
(All information on this sheet is contained in the block from Cell A1 to Cell I25)							
Required spacecraft power - sunlight		420.0	W	Total required solar power		897.1	W
Required spacecraft power - eclipse		420.0	W	Controlled spacecraft power		420.0	W
				Converted spacecraft power		420.0	W
Orbit period		101.17	min				
Maximum eclipse time		35.11	min	Ideal solar cell performance		386.9	W/m^2
Mission duration	2.000	2.000	yrs	BOL power capability		273.2	W/m^2
				EOL power capability		253.1	W/m^2
Solar flux		1367.0	W/m^2				
Worst-case Sun incidence angle		23.50	deg	Required solar array area		3.54	m^2
Transmission efficiency - sunlight		80.0%					
Transmission efficiency - eclipse		60.0%		Solar Array Mass & Power Budgets			
						Mass (kg)	Power (W)
Ideal solar cell efficiency	28.3%	28.3%					
Inherent degradation		77.0%		Solar Arrays			
Solar cell degradation per year		3.75%		Deployed		7.5	
Lifetime degradation		92.6%		Cylindrical, body-mounted		23.5	
				Omnidirectional, body-mounted		29.9	
Solar array power density	120.0	120.0	W/kg	Power Control Unit		8.4	
Spacecraft dry mass		34.6	kg	Regulator/Converters		10.5	84.0
Percent of spacecraft dry mass for wiring		4.0%		Wiring		1.4	44.9
Percent of spacecraft power for wiring		5.0%					

Table 20: Battery Design (17)

Return to Navigator	Power Subsystem - Secondary Battery Sizing					
(All information on this sheet is contained in the block from Cell A1 to Cell H14)						
Orbit period		101.17	min			
Maximum eclipse time		35.11	min			
Mission duration	2.000	2.000	years			
Required power during eclipse		420.0	W			
Transmission efficiency		90.0%				
Number of charge-discharge cycles		10398				
Depth of discharge		55.0%		Battery capacity	496.8	W-hr
				Battery capacity	17.7	A-hr
Energy density	153.5	153.5	W-hr/kg			
Bus voltage		28.0	V	Mass of batteries	3.2	kg

7.3.4 Propulsion

The next subsystem to be examined was propulsion. The biggest candidates were traditional hydrazine, green propulsion and electric propulsion. Hydrazine has well established flight history earning it a TRL of 9 and has plenty of thrust and ISP. Green propulsion could be used to promote safe long term storage and SV/LV integration since hydrazine is extremely hazardous, but the technology is TRL 6 so the technology will require additional development time. Electric propulsion was the third candidate, but its thrust is too small to be practical for minimizing time for constellation deployment.

Specs taken from the NASA report model a standard hydrazine engine with an ISP of 235 sec and thrust of 30.7 N was used in the Table 21 (20).

Table 21: Propulsion Design (17)

Chemical Propulsion System						
Specific impulse	235.0	235.0	sec	Minimum feasible specific impulse	13.5	sec
Spacecraft dry mass (excluding propulsion system)		33.4	kg	Initial stage mass	38.8	kg
Required Delta-V		251.3	m/s	Final stage mass	34.2	kg
Delta-V margin percentage		15.0%				
				Propellant mass	4.6	kg
Inert mass fraction		15.0%		Inert mass	0.8	kg
Initial thrust-to-weight ratio	0.80	0.80		Thrust	303.8	N
				Mass flow rate	0.1	kg/s

Return to Navigator	Propulsion System - Storage and Feed					
(All information on this sheet is contained in the block from Cell A1 to Cell H17)						
Thrust	30.7	30.7	N	Propellant flow rate	0.01	kg/s
Specific impulse	235.0	235.0	sec	Fuel flow rate	0.00	kg/s
Propellant mass		4.6	kg	Oxidizer flow rate	0.01	kg/s
Oxidizer to fuel ratio		3.50		Fuel mass	1.0	kg
Fuel density			kg/m^3	Fuel tank volume		m^3
Oxidizer density			kg/m^3	Radius of spherical fuel tank		m
Ullage fraction		3.00%		Oxidizer mass	3.6	kg
				Oxidizer tank volume		m^3
				Radius of spherical oxidizer tank		m
				Bulk density		kg/m^3
				Bulk volume		m^3

7.3.5 Structures

For structures, an analysis of a monocoque structure was conducted. Monocoque structures are used in CubeSats over semi-monocoque structures to increase the available space to fit in components. 6061-T6 aluminum was selected as it is a standard material for Cubesat frames. In the NASA small satellite state of the art report (20), the largest Cubesat frame size examined was 12U (TRL 7), so having a frame that can support the 27U sized SV is the riskiest component with a TRL of approximately 3 or 4 (20). According to Figure 20, the 27U frame has dimensions of 34 cm x 35 cm x 36 cm (18). Since the SMAD spreadsheet (Table 22) approximates the frame as a cylinder, rough dimensions that fit the 27U size were input with a frame thickness of 1 mm. The resulting factor of safety gives plenty of margin, however further research is required since the frame has not yet been validated in an operational environment. Examples of what the structure may look like are in Figures 19 and 20.

Table 22: Structures Analysis (17)

Return to Navigator	**Structural Analysis: Monocoque Structure**								
(All information on this sheet is contained in the block from Cell A1 to Cell O27)									
Cylinder length	*0.36*	0.36	m	***Rigidity (Stiffness)***					
Cylinder radius	*0.18*	0.18	m	Natural frequency (axial)		15.0	582.1	Hz	Set to value
Spacecraft mass		39.4	kg	Natural frequency (lateral)		10.0	461.0	Hz	Set to value
				Axial deflection			0.001	cm	
Young's modulus	*6.800E+10*	6.800E+10	N/m^2	Lateral deflection			0.000	cm	
Material density	*2.710E+03*	2.710E+03	kg/m^3						
Ultimate tensile stress	*2.900E+08*	2.900E+08	N/m^2	***Stability***					
Yield tensile stress	*2.400E+08*	2.400E+08	N/m^2	Ultimate load			4.826E+03	N	
Factor of safety (ultimate)		1.25		Geometric parameter (phi)			0.839		
Factor of safety (yield)		1.10		Reduction factor (gamma)			0.489		
				Buckling stress			1.107E+08	N/m^2	
Axial acceleration load factor		6.0	g's	Critical buckling load			1.252E+05	N	
Lateral acceleration load factor		2.0	g's	Margin of safety			24.95		Set to zero
				Applied loads					
Cylinder thickness	*1.000E-01*	1.000E-01	cm	Axial load			2.316E+03	N	
Cross-sectional area		1.131E+01	cm^2	Bending moment			1.390E+02	N-m	
Area moment of inertia		1.832E+03	cm^4	Equivalent axial load			3.861E+03	N	
				Ultimate load			4.826E+03	N	
Mass Budget				Yield load			4.247E+03	N	
Skin		1.1	kg	Margin of safety (ultimate)			66.97		Set to zero
Fasteners & fittings		0.1		Margin of safety (yield)			62.92		Set to zero
Total		1.2	kg						

Figure 19: Cubesat Sizes (18)

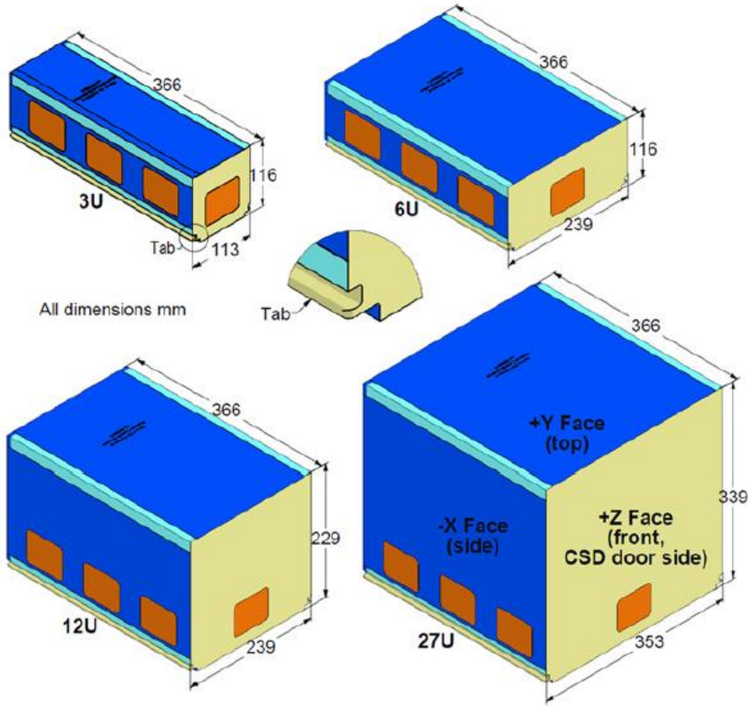
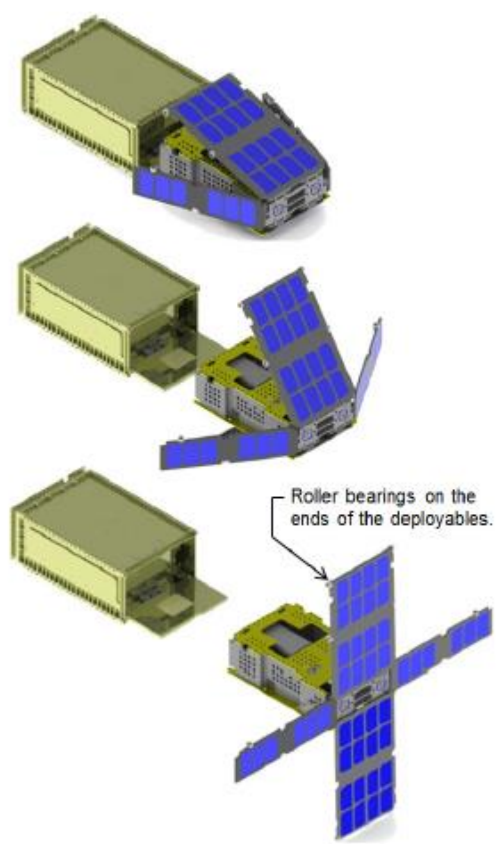


Figure 20: Example of 6U Cubesat Deployment (18)



7.3.6 Thermal

The final subsystem to be examined was thermal control. According to NASA, plenty of TRL 9 applications such as paint, MLI materials and thermal louvres exist to choose from (20). A surface area of 0.73 m^2 was input into the SMAD spreadsheet (Table 23) to correlate with the 27U frame size.

Table 23: Thermal Design (17)

Return to Navigator	Thermal Control Subsystem Analysis						
	(All information on this sheet is contained in the block from Cell A1 to Cell I30)						
	[NOTE: References on pp 445-447 refer to the 4th Printing of the 3rd Edition of SMAD]						
Orbit altitude		808.529	km	Solar flux		1418.0	W/m^2
Planet angular radius		62.56	deg	Albedo		34.0%	
Albedo reflection factor		0.991		Maximum Planet IR emission		258.0	W/m^2
				Minimum Planet IR emission		216.0	W/m^2
Available surface area	0.730	0.730	m^2				
Diameter of equivalent sphere		0.48	m	Solar energy absorbed		155.3	W
				Albedo energy absorbed		41.2	W
Absorptivity of spacecraft surface		60.00%		Maximum Planet IR energy absorbed		29.7	W
Emissivity of spacecraft surface		80.00%		Minimum Planet IR energy absorbed		24.8	W
Maximum power dissipation on spacecraft		428.5	W	Maximum equilibrium temperature		102.0	deg, C
Minimum power dissipation on spacecraft		428.5	W	Minimum equilibrium temperature		69.1	deg, C
Upper temperature limit for spacecraft		35.0	deg, C				
Lower temperature limit for spacecraft		5.0	deg, C				
Possible changes to reduce maximum equilibrium temperature to specified upper limit:							
Additional surface area		3.622	m^2	New absorptivity of spacecraft surface		-36.10%	
				New emissivity of spacecraft surface		92.50%	
Heater requirements during eclipse:							
Radiator area to accommodate s/c power dissipation		1.050	m^2	Maximum eclipse time		35.1	min
Minimum temperature for given radiator area		35.0	deg, C	Duty cycle (per orbit period)		34.8%	
Required heater power (during eclipse)		0.0	W	Average heater power		0.0	W

7.3.7 Final System Sizing

After all the inputs were gathered to what one would expect for the space system, a final system sizing and cost analysis was conducted in Table 24. The final projected wet mass of the spacecraft came out to be 59.5 kg with 7.2 kg of margin included. Though without margin the spacecraft just fits into the 54 kg limit, technology is always getting lighter every year and since the spreadsheet was provided to SYS632 in 2015 it can be expected to meet mass margins. However, one significant driver is the mass of the propellent required to perform deorbiting maneuvers. Since Cubesat have a 25 year legal mission life, it is possible to cut out propellent so just enough remains at the end of its 2 year mission life to put the SV in an orbit low enough for atmospheric drag to slowly de-orbit the SV over the remaining 23 years.

Table 24: Final System Estimates (17)

Return to Navigator

System Sizing Summary

(All information on this sheet is contained in the block from Cell A1 to Cell AN32)

Payload Mass	7.5	kg
Payload Percentage	27.0%	
Margin Percentage (Mass)	15.0%	
Payload Power (peak)	138.7	W
Payload Percentage	41.3%	
Margin Percentage (Power)	15.0%	

Type of attitude control system	Three-axis (zero momentum)
Type of communications	Separate uplink/downlink antennas
	Solid State Amplifier (SSPA)
Primary power source	Solar arrays - deployed
Type of propulsion system	Chemical
Type of structure	Monocoque

	Preliminary Estimates		Current Estimates		
	Mass (kg)	Avg Power (W)	Mass (kg)	Peak Power (W)	Avg Power (W)
Payload	7.5	138.7	7.5	138.7	138.7
S/C Subsystems	20.2	197.3	40.2	218.2	218.2
ADCS	*2.2*	*38.5*	*3.8*	*38.5*	*38.5*
C&DH	*1.2*	*15.9*	*1.2*	*15.9*	*15.9*
Power	*7.8*	*71.1*	*31.0*	*128.9*	*128.9*
Propulsion	*1.0*	*12.0*	*0.8*	*12.0*	*12.0*
Structure	*6.1*	*0.0*	*1.2*	*0.0*	*0.0*
Thermal	*1.0*	*12.0*	*1.0*	*0.0*	*0.0*
TT&C (Communications)	*0.9*	*47.8*	*1.3*	*22.9*	*22.9*
Margin	6.9	84.0	7.2	53.5	53.5
S/C Dry Mass	34.6		54.9		
Propellant Mass	4.8		4.6		
S/C Loaded Mass	39.4		59.5		
S/C Power		420.0		410.4	410.4

Size

	Minimum	Expected	Maximum	
Spacecraft				
Volume	*0.30*	**0.59**	*2.97*	m^3
Body area	*0.34*	**0.95**	*1.87*	m^2
Linear dim	*0.59*	**0.98**	*1.37*	m
Moment of inertia	--	**9.07**	--	kg-m^2

Solar Array		
Area	3.54	m^2
Mass	7.48	kg
Area offset (of deployed array)	2.13	m^2
Moments of inertia (of deployed array)		
Perpendicular to array face	36.11	kg-m^2
Perpendicular to array axis	35.01	kg-m^2
About array axis	1.10	kg-m^2

7.4 Cost Estimates

After the system was sized, a cost analysis was conducted in Table 25. For mission inputs, the payload was considered a communications payload because out of the three choices (comm, IR, visible light), comm fit the payload most closely. According to bSpace Launch, a provider of CubeSat hosted load slots, a 12U hosted load will cost $945,000 and a 27u is custom priced. Extrapolating the 12U cost to 27U led to a launch estimation of $2.1M per Cubesat (35), which is significant considering the cost of a ULA Delta IV or Atlas V is about $73M (36). However, note that the launch figure is a gray area since hosted loads may not be the quickest way to space, may not be the final selected launch system for this constellation and the bSpace website is not functional as of 15 Apr 2019. Five spacecraft were purchased to meet initial requirements. For flight heritage, all components were considered "basically existing design" while the S/C bus and Structure were considered "nominal new design". Payload was considered "moderate modifications to existing design". After the inputs were inserted into the SMAD sheet, final cost estimates using different methods were conducted with the worst case being $122M. Since the price of a single GPS III satellite is estimated at $577M (37), this space segment is 20% the cost. However, with small SV mass production initiatives such as the Airbus high volume satellite factory, cost estimates can be expected to dramatically decrease in the near future (23). For the price of one GPS III SV and the accompanied LV cost, a space segment with at least five hours of full CONUS coverage can be bought.

Table 25: Cost Estimates (17)

Return to Navigator

System Inputs for Cost Estimation

(All information on this sheet is contained in the block from Cell A1 to Cell AD28)

Mass Estimates (with margin)			
Payload mass		8.6	kg
Spacecraft bus dry mass		46.3	kg
ADCS		4.3	kg
C&DH		1.4	kg
Power		35.6	kg
Propulsion		0.9	kg
Structure		1.4	kg
Thermal		1.2	kg
TT&C		1.5	kg
Propellant mass		4.6	kg
Physical Dimension Estimates			
Spacecraft volume		0.595	m^3
Solar array area		3.545	m^2
Aperture diameter			m
Power Estimates			
Payload power (with margin)		159.5	W
BOL power		968.4	W
EOL power		897.1	W
Average power		410.4	W
Battery capacity		17.7	A-hr

Other Spacecraft Information			
Type of payload	Communications		
Type of attitude control	Three-axis		
Pointing accuracy			deg
Pointing knowledge			deg
Number of thrusters		1	
Data storage capacity			Mb
Downlink data rate		1250.00	Kbps
Number of spacecraft	5	5	
Launch Information			
Number of launches		5	
Cost per launch	2.1	2.1	$M
Operations Information			
Mission duration	2	2	yrs
Number of FTEs		105	
FTE - burdened rate		160.0	$K
Learning curve slope		95.0%	

Heritage	
Payload	Moderate modifications to existing design
S/C Bus	Nominal new design
ADCS	Basically existing design
C&DH	Basically existing design
Power	Basically existing design
Propulsion	Basically existing design
Structure	Nominal new design
Thermal	Basically existing design
TT&C	Basically existing design

Return to Navigator	Lifecycle Cost Estimate			
	(All information on this sheet is contained in the block from Cell A1 to Cell J22)			
	USCM 7th Edition (FY00 - $M)	SSCM (FY00 - $M)	SSCM (Ver 7.4) (FY00 - $M)	SSCM (Ver 8.0) (FY00 - $M)
Space Segment Cost				
using "S/C Bus - Total" estimate	$36.104	$26.768	$43.255	$80.353
using "S/C Bus - Sum of Components" estimate	$41.349	$41.456	$43.255	$80.353
using weighted average of S/C Bus estimates	$36.295	$33.306	$43.255	$80.353
Launch Costs	$10.500	$10.500	$10.500	$10.500
Operations and Maintenance Costs				
First Year	*$16.800*	*$16.800*	*$16.800*	*$16.800*
Lifetime	$31.920	$31.920	$31.920	$31.920
Total Lifecycle Cost				
using "S/C Bus - Total" estimate for Space Segment Costs	**$78.524**	**$69.188**	**$85.675**	**$122.773**
using "S/C Bus - Sum of Components" estimate for Space Segment Costs	**$83.769**	**$83.876**	**$85.675**	**$122.773**
using weighted average of S/C Bus estimates for Space Segment Costs	**$78.715**	**$75.726**	**$85.675**	**$122.773**

8.0 Discussion

The goal of this paper was to provide a top level analysis of the space segment architecture for a rapidly deployed GPS recovery constellation. The resulting estimates show that using a 27U Cubesat to provide GPS L1 Signal in LEO is a completely viable mission in terms of cost and technology availability. The biggest technological hurdle is the structure, however there is a marker for smaller as well as larger satellites so I see no project ending technical constraint that will prevent development. It is also always possible that atomic clock technology will continue to develop to result in smaller clocks capable providing accuracy worthy of the GPS mission that will drive down the proposed SV size and therefore technical risk. To perform further research on small GPS satellites from this paper, the next step in the Systems Engineering process would be to begin detailed SV design from this paper's architecture analysis. Though this paper provides an idea of how the subsystems will be sized, it does not perform the deep technical analysis required to design a spacecraft. Specific dimensions, interfaces, budgets and components will all have to undergo further technical scrutiny in order to yield a space-worthy vehicle. However, ultimately development of a small GPS satellite will be an exercise is systems integration to incorporate existing technology into a brand new configuration.

As noted in the research, a proper analysis of alternatives on the launch system needs to be accomplished. Alternatives that exist include hosted loads on traditional launch vehicles, using air launch systems and using dedicated rockets ready to go in a silo. Currently the use of air launch vehicles shows the most promise for this application. Launching a satellite from an aircraft has been done before using the Pegasus launch vehicle, however the Pegasus made its last flight in 2016. One could argue that this was because the launch capability emerged before there was a significant market need from the space segment, but that can be a whole paper by itself. Today, new specialized airframes are emerging in a market that is projected to reach a value of $7B by 2024 (23). On April 13th, 2019 the Stratolaunch Aircraft made its first flight (see Figure 21). Dubbed "the world's largest airplane" due to its football field sized wingspan, this behemoth is designed to carry rockets that insert small satellites into LEO (24). Another breaking air launch application is Virgin Orbit's Launcher One, a rocket designed to fit under a modified 747-400 dubbed "Cosmic Girl". Launcher One's first orbital flight is scheduled for Q2 of 2019 (see Figure 22) (25). Air launch technologies show great potential when it comes to rapidly deploying small satellites into LEO and it can be expected that these aircraft will become operationally available within a year or two. Due to the emergence of commercia air satellite launch, it is difficult to estimate just how much this solution will be in terms of launch cost. Therefore, proper launch vehicle analysis and comparison must be done before a selection can be made.

The next area of further research that would need to be accomplished is the capacity of the ground system to support the constellation. Considering the Next Generation Operational Control Segment (OCX), a historically troubled program that survived a Nunn-McCurdy breach, adding the paper's proposed small satellites into the GPS constellation may be become problematic. The system itself won't even be fully available for the next few years so it is not currently 100% operationally validated. Examination of the existing operational control segment may have to be done to assess whether these SVs are supportable. It may turn out that this paper's proposed contingency constellation cannot be integrated into any existing GPS ground control segment and modifications may have to be made for the system to work. Further analysis into the ground segment is required because the space segment will be dead in space without it.

The final of research I would recommend for this application is categorizing the different types of constellations required to fix different types of coverage gaps. There are multitudes of combinations of gaps that can occur, heaven forbid, if we start to lose GPS satellites prematurely. This research focused on the hypothetical situation where enough satellites were lost that only three had CONUS Line of Sight (LOS) for the duration of 1 hour. Since there are at least six GPS satellites in CONUS LOS at one time and that the constellation size was recently increased from twenty four to twenty seven SVs, this situation would require the loss of more than one existing GPS SV (39). But what if we lose more SVs leading to two, one or even no GPS SVs in LOS of the CONUS? What if we lose GPS coverage over a non-CONUS area where US military forces are operating? This additional research will also drive cost estimates for the program as it may include probability of losing GPS SVs prematurely to determine how many GPS small satellites should be bought. Maybe it would make sense to only buy a handful, or maybe it would make sense to buy enough to insure the entire GPS constellation. There are a multitude of combinations of unexpected GPS coverage outages that can occur, none of which are good, therefore further thought into this subject must be conducted.

Figure 21: StratoLaunch on its Maiden Flight in the Mojave Desert 13 Apr 2019 (25)

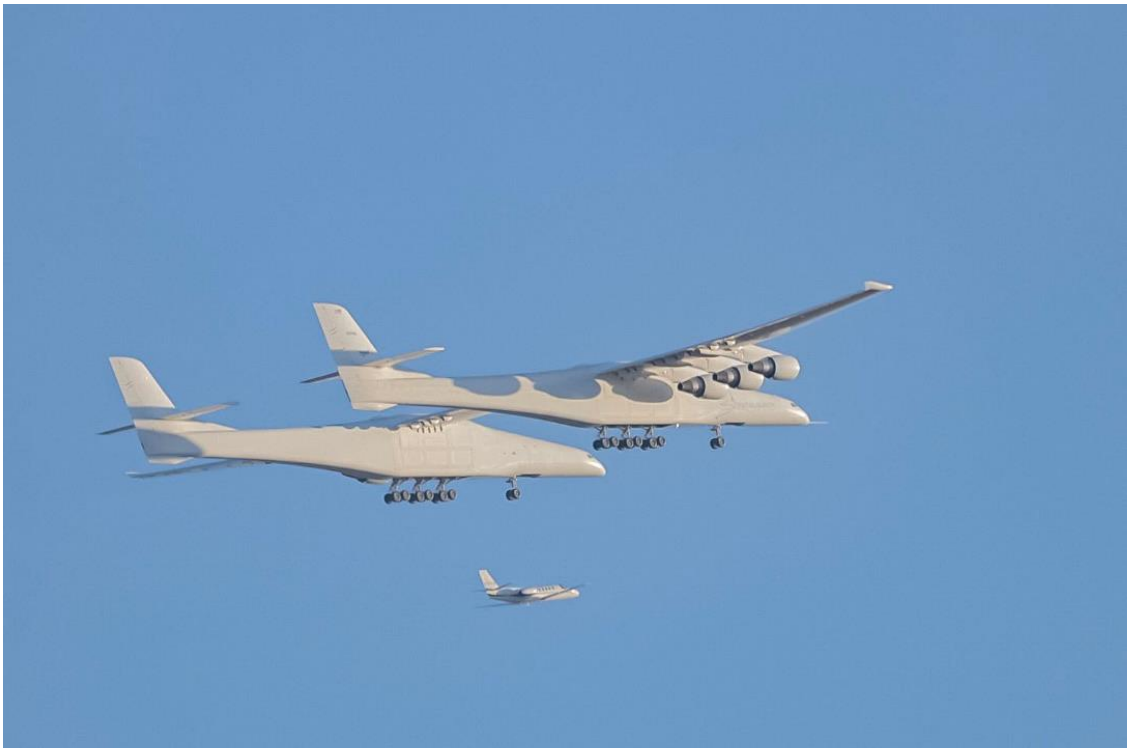


Figure 22: Virgin Orbit's Launcher One/Cosmic Girl Flight Test on 18 Nov 2018 (28)

9.0 Conclusion

The use of small satellites to rapidly fill a gap in GPS coverage looks as promising as the antisatellite threats to the US are real. The US needs to move faster than its adversaries to ensure space dominance and the use of small satellites can be the tip of the spear on that front. Acquire fast and launch fast is the way to go to beat out our adversaries in the space domain and this research provides a solution on how to do that. Small satellite technology is not limited to the avantgarde concept, back-up contingency platforms or GPS- it can be employed in mainstream space operations. The technology for small satellites is becoming more widely available and the market is certainly growing. In the future it may be entirely possible that small satellites comprise the physical architecture for GPS IV; time will tell. The United States has been used to operating uncontested in space, however with emerging space threats from other countries and rogue actors, it can never fall complacent if it is to retain its superiority in the space domain. Thank you for reading, I hope this research has been thought provoking.

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